GROUND STUDIES FOR PILOTS

FLIGHT INSTRUMENTS & AUTOMATIC FLIGHT CONTROL

SIXTH EDITION

DAVID HARRIS

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Sixth Edition

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Preface

Since the publication of Roy Underdown's Fifth Edition of Volume 3 of *Ground Studies for Pilots*, the format of the JAR-FCL examinations has become established. Among other things, it has become clear that students need the materials assembled in such a way as to facilitate study for individual papers and that it is no longer practical to combine Instruments with Navigation General, for example. The scope of what was once called the Instruments paper (now paper number 022 in the ATPL syllabus) is now so broad that it justifies a volume in its own right. Whilst still covering the air data and gyroscopic instruments, inertial navigation systems and electronic navigation instruments, it has now been extended to cover the additional syllabus requirements of engine and systems monitoring instruments and flight warning systems.

As more and more students find the need to study subjects in the groupings as they appear in the examinations, it makes sense to group those subjects accordingly in the *Ground Studies for Pilots* series of books. This new Sixth Edition addresses all the subjects listed in the JAR Learning Objectives for the Instruments and Automatic Flight paper number 022. I have tried to make the book both readable and instructive, with the intention that it should be useful to those seeking general information as well as to the examination student. Whilst it is aimed principally at pilots studying for the JAR ATPL ground examinations, it will also be helpful to pilots at all professional and private levels.

Just a few years ago, automatic flight systems and electronic instrument systems were almost exclusively the preserve of large passenger transport aircraft. In more recent times, however, it has become increasingly common for smaller short-haul and executive-type jet and turbo-prop aircraft to be equipped with such systems. Consequently, there is a need for those pilots intending to progress from light and general aviation into commercial flying to have knowledge of at least the basic principles of these systems and instruments, even though they may not be immediately intending to sit the professional examinations. The text and diagrams in this volume have been deliberately designed to be understandable without preknowledge of the subjects.

Acknowledgement
The assistance of Roger Henshaw, Peter Swatton and David Webb, all of Ground Training Services Ltd at Bournemouth International Airport, is gratefully acknowledged for their advice and for scrutinising the manuscript of this book for errors. Their professional knowledge of the JAR-FCL examination requirements has helped to ensure that it will be extremely useful to pilots undertaking ground studies.

David Harris
Minehead

List of Abbreviations

a.c.	alternating current
ACARS	aircraft communications addressing and reporting system
ACAS	airborne collision avoidance system
ADC	air data computer
ADI	attitude director indicator
agl	above ground level
AIDS	aircraft integrated data system
amsl	above mean sea level
AOM	Aircraft Operating Manual
ASI	airspeed indicator
ASIR	airspeed indicator reading
BITE	built-in test equipment
CADC	central air data computer
CAS	calibrated airspeed
CDU	control and display unit
CEC	compressibility error correction
CG	centre of gravity
CHT	cylinder head temperature
COAT	corrected outside air temperature
CRT	cathode ray tube
CSDU	constant speed drive unit
CWS	control wheel steering
d.c.	direct current
DEC	density error correction
DG	directional gyro
DH	decision height
EADI	electronic attitude and direction indicator
EAS	equivalent airspeed
ECAM	Engine Centralised Aircraft Monitoring
EFIS	Electronic Flight Instrument System
EGT	exhaust gas temperature
EHSI	electronic horizontal situation indicator
EICAS	Engine Indicating and Crew Alerting System
EPR	engine pressure ratio

FFRATS	Full Flight Regime Autothrottle System
FL	Flight Levels
FMC	flight management computer
FMS	flight management system
FOG	fibre optic gyro
GPWS	Ground Proximity Warning System
HDG	heading
HSI	horizontal situation indicator
hPa	hectopascal
IAS	indicated airspeed
ICAO	International Civil Aviation Organisation
IE	instrument error
IEC	instrument error correction
in Hg	inches of mercury
INS	Inertial Navigation System
IRS	Inertial Reference System
ISA	International Standard Atmosphere
IVSI	instantaneous vertical speed indicator
JAA	Joint Aviation Authority
JAR	Joint Aviation Regulations
K	degrees kelvin
kg	kilogram(s)
lb	pound(s)
LCD	liquid crystal diode
LED	light emitting diode
LNAV	lateral navigation
LSS	local speed of sound
LVDT	linear voltage displacement transmitter
M	mach number
MAP	manifold air pressure
MCDU	multi-purpose control and display unit
M_{crit}	critical mach number
msl	mean sea level
nm	nautical miles
OAT	outside air temperature
P	pitot pressure
PE	position error
PEC	pressure error correction
Q	dynamic pressure
QFE	height above a chosen ground datum
QNH	height above mean sea level
RA	resolution advisory
RAS	rectified airspeed

RCDI	rate of climb/descent indicator
RLG	ring laser gyro
RMI	radio magnetic indicator
rpm	revolutions per minute
S	static pressure
SAT	static air temperature
SBY	standby
SSR	secondary surveillance radar
TA	traffic advisory
TAS	true airspeed
TAT	total air temperature
TAWS	Terrain Avoidance Warning System
TCAS	Traffic Collision Avoidance System
TCS	touch control steering
TOD	top-of-descent
V	airspeed
VNAV	vertical navigation
VOR	VHF omnidirectional ranging
VSI	vertical speed indicator

Chapter 1
Air Data Instruments

Two of the most important pieces of information for a safe flight are height and speed. Almost from the beginning of powered flight these have been provided to the pilot by instruments that utilise the ambient atmospheric pressure by means of a pitot/static system.

Pitot and static systems

Static pressure

The ambient atmospheric pressure at any location is known as the static pressure. This pressure, in a standard atmosphere, decreases by 1 hectopascal (hPa) for each 27 feet (ft) increase in altitude at mean sea level. For simplicity this figure is usually approximated to 1 hPa per 30 ft gain in altitude. The rate of change of pressure with height is fundamental to the operation of the pressure altimeter, the vertical speed indicator and the mach meter. Each of these instruments uses static pressure to measure aircraft altitude, or rate of change of altitude.

Static pressure, that is the pressure of the stationary air surrounding an aircraft, irrespective of its height or speed, is sensed through a set of small holes situated at a point on the aircraft unaffected by turbulence. This sensing point is known as the static source. It is typically on the side of the fuselage or on the side of a tube projecting into the airstream.

Pitot pressure

As an aircraft moves through the air it displaces the surrounding air. As it moves forward it compresses the air and there is a pressure increase on the forward-facing parts of the aircraft. This pressure is known as dynamic pressure.

Suppose a cup were to be placed on the front of an aircraft, with its open end facing forward. When the aircraft is stationary the pressure inside the cup will be the same as the surrounding air pressure. In other words it will be static pressure. When the aircraft begins to move forward the air inside the cup will be compressed and dynamic pressure will be added to the static

pressure. The faster the aircraft moves, the greater the dynamic pressure will become, but static pressure will always also be present.

The pressure measured on the forward-facing surfaces of an aircraft will be the sum of static pressure and dynamic pressure. This is known as *pitot pressure*, or total pressure, which is sensed by a forward-facing, open-ended tube called a pitot tube, or pitot head. Figure 1.1 is a simplified diagram of a pitot head.

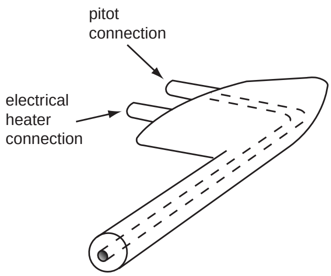

Figure 1.1 Pitot head.

The pitot head comprises an aerodynamically shaped casing, usually mounted beneath one wing, or on the side of the forward fuselage, clear of any turbulent airflow. Within the casing is a tube, the rear of which is connected to the pitot system, which conveys pitot pressure to the pilot's instruments. An electrical heating element is fitted within the tube to prevent the formation of ice, which could otherwise block the tube and render it useless. Drain holes are provided in the bottom of the tube to allow water to escape.

The dynamic element of pitot pressure is required to operate those air data instruments that display speed relative to the surrounding air, the airspeed indicator and the mach meter.

Pitot and static pressure is supplied to the air data instruments through a system of tubes known as the pitot/static system. A schematic layout of the pitot/static system for a light aircraft is shown in Figure 1.2. The static source is duplicated on either side of the rear fuselage. This is to compensate for false readings that would occur if the aircraft were side-slipping or in a crosswind. The pitot and static pressure supplies are connected to a duplicate set of instruments in aircraft that have a pilot and co-pilot. In many aircraft the pitot and static sources are combined in the pitot head, as illustrated in Figure 1.3.

The static source consists of a number of small holes in the side of the pitot head, connected to an annular chamber surrounding the pitot tube. This chamber is connected to the static system, which conveys static pressure to the pilot's instruments. A separate pipe connects the pitot tube to the pitot

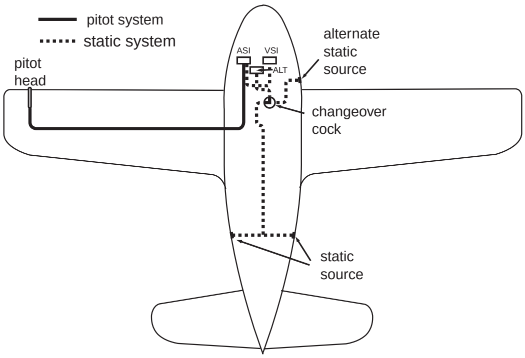

Figure 1.2 Pitot/static system for a light aircraft.

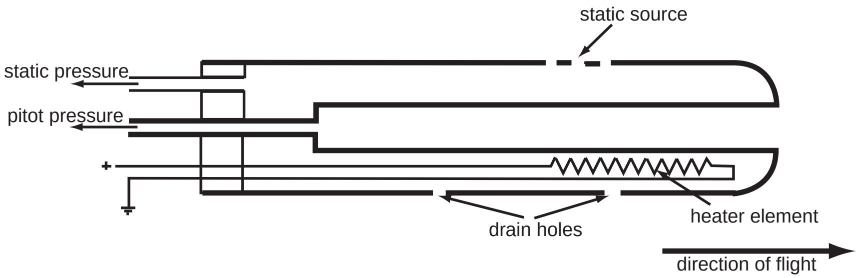

Figure 1.3 Pitot/static head.

system. As with the pitot head shown in Figure 1.1, an electrical heating element is fitted to prevent blockage of the pitot and static sources due to icing and water drain holes are provided in the bottom of the casing. In some aircraft this type of pitot head is mounted on the fuselage near the nose and it may be duplicated, one each side, to compensate for crosswind effects.

Static pressure error

The static system of air pressure measurement will be incorrect if the airflow is turbulent, if there is a crosswind, or if the aircraft is side-slipping. The effect of turbulence is minimised by locating the static source clear of protuberances and disturbed airflow. To eliminate the effect of crosswind or side-slip the static source is duplicated and this is known as static balancing. Despite all the measures taken by the aircraft designer there is often some small error in the sensed static pressure, but this can usually be measured and compensated for by a correction card or table. In many aircraft the correction values will differ according to the position of flaps and/or landing gear.

Malfunctions

- Correct and reliable indications from the various air data instruments can only be achieved if the pitot and static sources are kept clear of any blockages and the pitot/static system within the aircraft remains undamaged and pressure-tight.
- Blockage of the pitot or static source may occur due to icing, insects, dirt or dust. It is also not unknown for aircraft painters to forget to remove masking tape from the perforated discs that form the static vents on the fuselage sides. Icing can be prevented by the use of heaters, but this may affect the sensed pressure to some small extent. The effect of blockages is to render the instruments dangerously inaccurate or useless.
- Blockage of the static sources will cause the altimeter reading to remain constant regardless of changing aircraft altitude, the vertical speed indicator will not indicate rate of change of height and the airspeed indicator will be dangerously inaccurate.
- Blockage of the pitot source will not affect the altimeter or vertical speed indicator, but it will render the airspeed indicator useless and the mach meter grossly inaccurate.
- Leakage in the piping of the pitot/static system will also seriously affect the accuracy and usefulness of the air data instruments. Loss of pitot pressure due to leakage in the pitot pressure system will cause the airspeed indicator to underread.
- Leakage in the static pressure system within the cabin of a pressurised aircraft is a serious problem, since the altimeter will register an altitude equivalent to cabin altitude, which will almost certainly be much lower than aircraft altitude. The vertical speed indicator will not function at all and the airspeed indicator will be inaccurate.
- In unpressurised aircraft the effect of a static system leak is less serious, since internal pressure is much the same as external. However, it may change at a slightly slower rate when the aircraft is climbing or descending and this would clearly affect the accuracy of the pressure instruments during height changes.

Alternate static source

Blockage of the static source is a more probable hazard in flight and for this reason many aircraft are fitted with an alternate static source. This may take pressure from within the cabin in the case of unpressurised aircraft, or from a separate external source. In either case there is likely to be a slight difference in pressure compared with that from the normal source. The aircraft flight manual usually contains correction values to be used when the alter-

nate static source has been selected. Changeover is made by means of a selector cock easily accessible to the pilot.

In light aircraft, not fitted with an alternate static source, if the static source is blocked an alternative source can be obtained by breaking the glass of the vertical speed indicator (VSI).

The pressure altimeter

The function of the pressure altimeter is to indicate the aircraft height above a given pressure datum. It operates on the principle of decreasing atmospheric pressure with increasing height and is, in fact, simply an aneroid barometer that is calibrated to read pressure in terms of height. To do this, the manufacturer assumes that air pressure changes at a given rate with change of height. The International Standard Atmosphere (ISA) values are the data used for this assumption.

In the ISA the temperature at mean sea level is +15°C and the air pressure is 1013.25 hectopascals (hPa). The temperature lapse rate (the rate at which the temperature will decrease with increase of height) is 1.98°C per 1000 ft (6.5°C per kilometre) up to a height of 36 090 ft. Above that height the temperature is assumed to remain constant at −56.5°C up to a height of 65 600 ft.

Air is a fluid and it has mass, and therefore density. If we consider a column of air, its mass exerts pressure at the base of the column; the taller the column the greater the pressure exerted at the base. At any given height in the column the pressure exerted is proportional to the mass of air above that point and is known as hydrostatic pressure. Atmospheric pressure is assumed to decrease at a rate of 1 hPa per 27 ft gain in height at sea level, this rate decreases as height increases, so that at a height of say 5500 metres (18 000 ft) the same 1 hPa change is equivalent to approximately 15 metres (50 ft) change in height.

The pressure altimeter is calibrated to read height above a selected pressure datum for any specific atmospheric pressure.

The element of the pressure altimeter that measures atmospheric pressure changes is a sealed capsule made from thin metal sheet. The capsule is partially evacuated, so that the surrounding atmospheric pressure tends to compress the capsule. However, a leaf spring attached to the capsule prevents this. The capsule may be of the diaphragm or the bellows type, as illustrated in Figure 1.4.

The simple altimeter

The capsule is mounted in a sealed casing, connected to the static source. Increased static pressure will cause the capsule to be compressed against the restraining force of the leaf spring, decreased static pressure will allow the

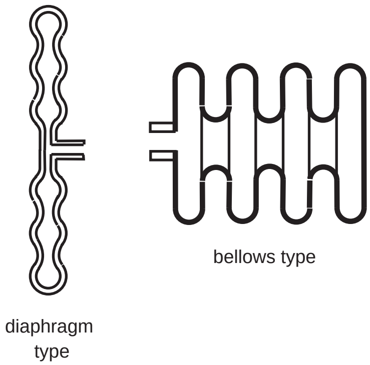

Figure 1.4 Aneroid capsule types.

leaf spring to expand the capsule. The spring force ensures that the extent of compression or expansion is proportional to the static pressure being measured. This compression or expansion of the capsule is converted into rotary motion of a pointer against a calibrated scale by a system of linkages and gears. A schematic diagram of a simple pressure altimeter is shown in Figure 1.5. Expansion of the aneroid capsule will cause a lever to pivot about its attachment to the instrument casing. This lever is connected to a drum by means of a chain and its pivoting motion causes the drum to rotate. The drum is attached to a pointer, which will consequently rotate against a calibrated card scale.

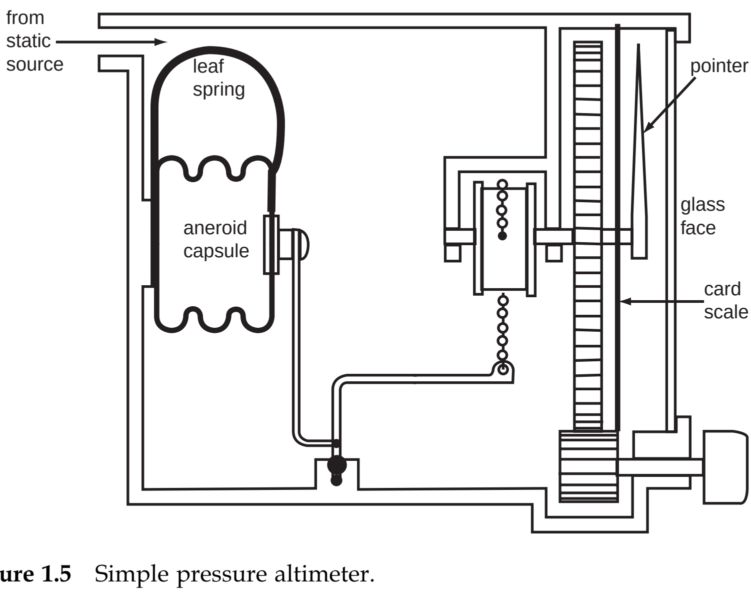

Figure 1.5 Simple pressure altimeter.

The setting of the height-indicating pointer can be adjusted by means of the pointer setting knob. The purpose of this is to allow the pilot to set the altimeter so that it displays height above a chosen datum. For example, if the pointer is set to read zero with the aircraft on the airfield the altimeter will thereafter indicate height above the airfield. Alternatively if, when the aircraft is on the airfield, the pointer is set to read height above mean sea level it will thereafter show height above mean sea level. In both these examples the surface pressure is assumed to remain constant.

The pressure altimeter is calibrated to read height in feet above the ISA mean sea level pressure of 1013.25 hPa. Some altimeters of US manufacture use inches of mercury (in Hg) as the unit of calibration and the equivalent of 1013.25 hPa is 29.92 in Hg. To convert inches Hg to hPa it is necessary to multiply by 33.8639. Conversely, to convert hPa to inches Hg it is necessary to multiply by 0.02953.

Some simple altimeters are calibrated on the assumption that static pressure reduces with increasing height at a constant rate of 1 hPa per 30 ft. Whilst this is a reasonable approximation up to about 10 000 ft, the height increase represented by 1 hPa decrease in pressure becomes progressively greater above this altitude. This is indicated in Table 1.1.

Table 1.1 Rate of pressure change with altitude.

Height in ISA (ft)	**Static pressure (hPa)**	**Height change (ft) represented by 1 hPa change in static pressure**
Mean sea level	1013.25	27
10 000	697	34
20 000	466	47
40 000	186	96
60 000	72	263

In order to have a linear scale on the altimeter display it is necessary to compensate for this non-linear distribution of atmospheric pressure. This is achieved during manufacture by careful design of the capsules and the mechanical linkages.

The simple altimeter is rarely used in aircraft because it lacks sensitivity, furthermore the altitude scale is compressed and causes confusion. More accurate height measurement is achieved with the sensitive altimeter.

The sensitive altimeter

The principle of operation of the sensitive altimeter is essentially the same as

that of the simple altimeter. Sensitivity of response to static pressure changes is improved by incorporating two or three aneroid capsules connected in series with each other. In a typical instrument the movement of the capsules as altitude changes is transmitted to gearing via a rocking shaft. The gearing in turn operates the height-indicating pointers, of which there are two or three.

Clearly, as an aircraft climbs the air temperature will fall. The altimeter is calibrated to be accurate when operating at a sea level air temperature of +15°C (ISA mean sea level temperature) and without compensation it will become increasingly inaccurate as instrument temperature deviates from that datum value. In the sensitive altimeter, temperature compensation is effected by a bimetallic strip inserted between the aneroid capsules and the transmission shaft. Note that this temperature compensation has nothing to do with altitude temperature error, which is discussed in sub-paragraph (6) under *Altimeter errors* below.

Figure 1.6 illustrates the display of a sensitive altimeter with three pointers.

It is perhaps easiest to think of the display as being similar to a clock face, in which one revolution of the long hand takes one hour and one revolution of the shorter hand takes 12 hours. In the altimeter display one revolution of the longest pointer represents a height change of 1000 ft, thus each numbered division on the instrument scale represents a height increment of 100 ft when read against this pointer.

One revolution of the medium length pointer represents a height change of 10 000 ft, thus each numbered division on the instrument scale represents a height increment of 1000 ft when read against this pointer. The height scale is divided into ten equal parts, so for every complete revolution of the 1000-ft pointer the 10 000-ft pointer moves one-tenth of a revolution.

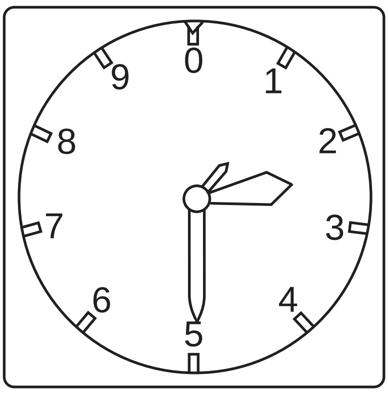

Figure 1.6 Sensitive altimeter display.

The shortest pointer shows increments of 10 000 ft for each numbered division on the instrument scale. The sensitive altimeter display in Figure 1.6 is therefore indicating a pressure altitude of 12 500 ft.

Whilst the sensitive altimeter is usable to greater altitudes than the simple altimeter it becomes inaccurate at high altitudes where the pressure change becomes small for a given increase in height. Its greatest disadvantage, however, is its multi-pointer display, which is very open to misinterpretation.

Subscale settings

Sensitive altimeters incorporate a pressure datum adjustment, so that height above any desired datum pressure will be indicated. The selected datum pressure is indicated on a subscale calibrated in hPa or in Hg, and this is usually referred to as the subscale setting.

QFE

If airfield datum pressure is set on the subscale the altimeter will indicate height above the airfield once the aircraft is airborne and zero when on the ground at that airfield. This setting is assigned the ICAO code of QFE for communication purposes.

QNH

If mean sea level pressure is set on the subscale the altimeter will indicate height above mean sea level. Thus, when the aircraft is on the airfield the altimeter will indicate airfield elevation above mean sea level. This setting is assigned the ICAO code of QNH for communication purposes.

Alternatively, the subscale may be set to 1013 hPa (29.92 in Hg), in which case the altimeter will indicate pressure altitude, that is the altitude above this pressure datum. This setting is used for aircraft flying Flight Levels (FL).

At this point it is appropriate to define the terms height and altitude with reference to altimeter subscale settings.

Height

Height is the vertical distance above a specified datum with known elevation, such as an airfield. Hence, if QFE is set on an altimeter it will indicate height above that airfield.

Altitude

Altitude is the vertical distance above mean sea level and it is therefore altitude that is indicated by an altimeter with QNH set. Pressure altitude, as we have already seen, is the altitude indicated on an altimeter with 1013 hPa, or 29.92 in Hg, set on its subscale.

Density altitude is the height in ISA conditions at which a given air density will occur and it is temperature dependent. If the local air temperature is ISA temperature, then density altitude will be the same as pressure altitude. If local air temperature is higher than ISA, then density altitude will be higher than pressure altitude and vice versa.

True altitude is the exact vertical distance above any point on the surface. A pressure altimeter cannot be relied upon to show true altitude, even with QFE set, since its calibration assumes a rate of pressure and temperature change with height that may not exist in the prevailing atmospheric conditions.

The servo-assisted altimeter

Whereas in the simple and sensitive altimeter the movement of the aneroid capsule directly moves the altimeter pointer, in the servo-assisted altimeter the capsule movement is transmitted to the indicator by an electro-mechanical system. When the capsules expand or contract their movement causes an electro-magnetic transducer to transmit an electrical signal to a motor, which drives the altimeter's single pointer and its height counters.

The servo-assisted altimeter has several advantages. Since the only mechanical transmission is from the capsules to the transducer there is considerably less resistance to motion and therefore less lag (see *Lag error* below). It is much more sensitive to small pressure changes and therefore remains accurate to greater altitude, where the pressure change is less for a given change of height. Because its output is electrical it can readily be used in conjunction with remote displays, altitude alerting and height encoding systems. The servo-assisted altimeter display with its height counters, as shown in Figure 1.7, is less likely to be misinterpreted than that of the sensitive altimeter.

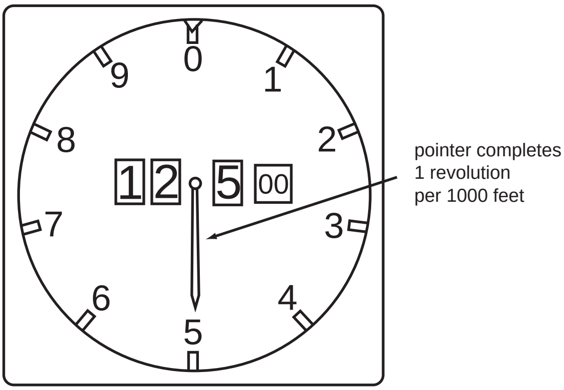

Figure 1.7 Servo-assisted altimeter display.

Altimeter errors

Pressure altimeters are prone to a number of errors, as listed below.

(1) **Lag error**. There is inevitably a delay between an atmospheric pressure change and the response of the aneroid capsules, with the result that the movement of the instrument pointer lags behind a change in altitude. The magnitude of the error depends upon the rate of change of altitude and is clearly unacceptable. In sensitive altimeters it is often reduced by means of a vibrator, or knocking, mechanism, which has the same effect as tapping a barometer to make the pointer move after a small pressure change. In servo-assisted altimeters lag error is virtually eliminated by the reduction of mechanical resistance.

(2) **Blockage of static source**. If the static source becomes blocked the altimeter will cease to indicate changes of static pressure, and therefore of altitude. The effect will be that the altimeter will continue to indicate the reading at which the blockage occurred.

(3) **Instrument error (IE)**. As with any mechanical device, the pressure altimeter is manufactured with small tolerances in its moving parts and these give rise to small inaccuracies in its performance. They are usually insignificant, but in some cases a correction table may be supplied with the instrument.

(4) **Position error (PE)**. The static source is positioned at a point on the airframe where disturbance to the airflow is minimal, so that the static pressure measured is as close as possible to the undisturbed ambient static pressure. However, there is usually some small error due to the positioning of the static source. Use of the alternate static source may also cause pressure error, since its siting is different to that of the normal source and thus subject to different effects during manoeuvring. The PE for an aircraft type is determined during its initial flight tests and is supplied in tabular or graphical format so that the pilot may make the appropriate corrections. In servo-assisted altimeters the correction is usually incorporated into the transducer system.

Corrections to be applied to the altimeter reading for position error may be listed in the Aircraft Operating Manual (AOM). These are usually small and take account of the effects of aircraft speed, weight, attitude and configuration, since all of these factors have some effect on airflow over the static source. Corrections are also usually given for both the normal and alternate static source, since position error may not be the same for both. The correction is applied as indicated in the AOM tables. For example, suppose the tables state that the correction to be applied with flaps at the landing setting of 45° is +25 ft, then 25 ft must be added to the altimeter reading.

(5) **Pressure error**. Also known as barometric error and subscale-setting error, this occurs when the actual datum pressure differs from that selected on the subscale setting of the altimeter. Suppose, for example, the subscale has been set for a regional QNH of 1020 hPa, but the aircraft is now operating in an area where the mean sea level (msl) pressure is 1000 hPa. Let us assume that the aircraft is at an altitude where the ambient pressure is 920 hPa, so the altimeter will indicate 3000 feet:

$(1020 - 920 = 100 \text{ hPa} \times 30 \text{ ft/hPa} = 3000 \text{ ft})$.

However, because of the lower msl pressure the actual height of the aircraft above msl is:

$1000 - 920 = 80 \text{ hPa} \times 30 \text{ ft/hPa} = 2400$ feet.

(6) **Temperature error**. The altimeter is calibrated to assume a Standard Atmosphere temperature lapse rate of 1.98°C per 1000 ft. Actual temperatures at any given altitude usually differ from this assumption and the altimeter will be in error. In cold air the density is greater than in warm air and a given pressure will occur at a lower altitude than in warmer air. Consequently, the altimeter will overread and, with QNH set, the aircraft will be lower than indicated, with the error increasing from zero at msl to a significant amount at altitude. The AOM contains a table of corrections for this situation. These corrections must be added to published or calculated heights/altitudes when temperature is less than ISA.

A 'rule of thumb' calculation of altitude temperature error is that the error will be approximately 4% of indicated altitude for every 10°C temperature deviation from ISA.

Altimeter tolerances

For altimeters with a test range of 0 to 9000 metres (0 to 30 000 ft) the required tolerance is ± 20 metres or ± 60 ft. For altimeters with a test range of 0 to 15 000 metres (0 to 50 000 ft) the required tolerance is ± 25 metres or ± 80 ft. The test is required to be carried out by the flight crew prior to flight, with QNH or QFE set and the altimeter vibrated either manually or by the mechanical vibration mechanism.

The airspeed indicator (ASI)

It is essential for the pilot of an aircraft to know its airspeed, because many critical factors depend upon the speed of flight. For example, the pilot needs

to know when the aircraft is moving fast enough for take-off, when it is flying close to the stalling speed, when it has accelerated to the speed at which landing gear and flaps must be raised and when it is approaching the maximum safe flying speed, to name but a few. This critical information is provided by the airspeed indicator (ASI).

The aircraft's speed relative to the surrounding air is proportional to the dynamic pressure that results from the air being brought to rest on forward facing parts of the airframe.

The airspeed indicator measures dynamic pressure and converts this to an indication of airspeed. We have already seen that pitot pressure (P), as measured in the pitot tube, is a combination of dynamic pressure (D) and static pressure (S), i.e. $P = D + S$. It therefore follows that dynamic pressure is pitot pressure less static pressure, i.e. $D = P - S$. Thus, the function of the ASI is to remove the static pressure element of pitot pressure and use the resulting dynamic pressure to move a pointer around a graduated scale.

The instrument comprises a sealed case connected to the static source and containing a capsule supplied with pitot pressure. Hence, the static pressure element of pitot pressure inside the capsule is balanced by static pressure surrounding the capsule. Consequently, the capsule will respond only to changes in the dynamic pressure element of pitot pressure. The faster the aircraft flies through the atmosphere, the greater will be the resultant dynamic pressure and the capsule will expand. This expansion is transmitted to a pointer by means of gearing and linkages. The principle is illustrated schematically in Figure 1.8.

A simple ASI typically uses a single pointer that moves around a scale calibrated in knots. More complex instruments may be used in high-speed aircraft, incorporating an angle-of-attack indicator or a mach meter.

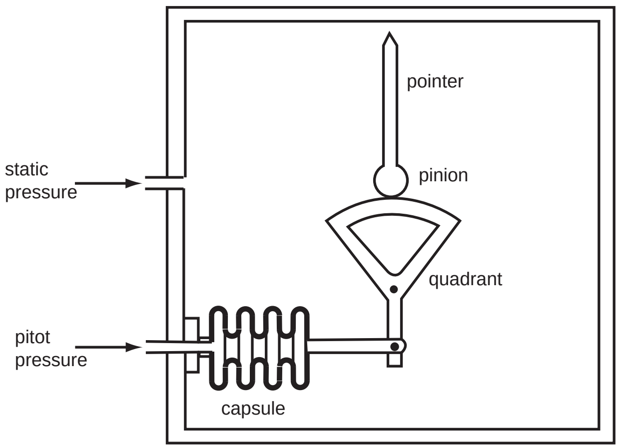

Figure 1.8 Principle of operation of the airspeed indicator.

Clearly the airspeed indicator must be calibrated to some fixed datum, for the pilot must be confident that an indication of, say, 90 knots is aerodynamically the same in all conditions. The datum used is the air density when ISA mean sea level conditions prevail, that is the density when the air temperature is +15°C and the air pressure is 1013 hPa. Only when these conditions prevail will the ASI indicate the true airspeed.

Any point where air is brought to rest, such as in the pitot tube, is known as a stagnation point. At this point the kinetic energy of the air is converted to pressure energy. The pressure resulting is, of course, dynamic pressure and is denoted by the symbol Q. It can be shown mathematically that $Q = \frac{1}{2}\rho V^2$, where ρ is the air density and V is the airspeed.

As the aircraft climbs the air density decreases and the airspeed indicator reading will be lower than the true speed of the aircraft through the surrounding atmosphere. A simplistic way of thinking of this is to consider that as the aircraft moves through the air it is colliding with the air molecules; the faster it flies the more molecules it strikes in a given time period.

The ASI indication of airspeed is based on the dynamic pressure measured, assuming an ISA mean sea level value of air density. Therefore, when the air density is lower, as with an increase of altitude, a given dynamic pressure (and therefore indicated airspeed) will only be achieved at a higher true airspeed.

Let us assume that it is flying in ISA msl conditions and is colliding with air molecules at a rate that causes the ASI to indicate 90 knots flight speed. When the aircraft climbs to a greater altitude the air density is less and so the molecules are further apart. In order for the ASI to continue to read 90 knots the aircraft must fly faster relative to the surrounding air in order to strike the same number of molecules within a given time period. Thus the true air speed, which is the speed of the aircraft relative to the surrounding atmosphere, increases.

Airspeed may be quoted in a number of ways and these are listed below:

ASIR: Airspeed indicator reading.

IAS: Indicated airspeed (IAS) is the speed indicated by the simple airspeed indicator reading (ASIR) corrected for errors due to manufacturing tolerances in the instrument (instrument error), but not corrected for static pressure errors occurring at the static source (pressure error).

CAS: Calibrated airspeed (CAS) is the airspeed obtained when the corrections for instrument error (IEC) and pressure error (PEC) have been applied to IAS. These correction values are usually found in the Aircraft Operating Manual and may be reproduced on a reference card kept in the cockpit. CAS = IAS + PEC

Calibrated airspeed used to be referred to as rectified airspeed (RAS).

EAS: Because air is a compressible fluid it tends to become compressed as it is brought to rest. At low to medium airspeeds the effect of compression is negligible, but above about 300 knots TAS the compression in the pitot tube is sufficient to cause the airspeed indicator to overread significantly. The greater the airspeed above this threshold, the greater the error due to compression. The effect is also increasingly noticeable at high altitude, where the lower density air is more easily compressed. The error produced by this effect is known as compressibility error.

Compressibility error correction (CEC) can be calculated and when applied to the calibrated airspeed the result is known as equivalent airspeed (EAS). EAS = CAS + CEC.

TAS: The airspeed indicator only indicates true airspeed (TAS) when ISA mean sea level conditions prevail; any change of air density from those conditions will cause the indicated airspeed to differ from true airspeed. The greater the altitude, the lower will be air density and therefore IAS (and consequently EAS) will be progressively lower than TAS.

The error due to the difference in density from ISA msl density can be calculated. When this density error correction (DEC) is applied to EAS the result is the aircraft's true airspeed (TAS). TAS = EAS + DEC. The compressibility error and density error corrections are embodied in the circular slide rule (navigation computer), from which TAS can be found using CAS and the appropriate altitude, speed and temperature settings.

Square law compensation

Given that dynamic pressure increases as the square of airspeed it follows that expansion of the capsule in the ASI must do likewise. Thus, at low airspeeds the amount of expansion for a given increase in speed will be small, whereas at higher airspeeds the amount of expansion will be relatively large for the same speed increase. This is known as square law expansion and, transmitted to the instrument pointer, will require an expanding scale on the face of the ASI, as illustrated at Figure 1.9. Such a scale makes accurate interpretation of airspeed difficult at low speeds and limits the upper speed range that can be displayed.

Most ASIs incorporate an internal compensation device that permits a linear scale on the dial, i.e. equal spacing of the speed divisions over the whole range, as depicted in Figure 1.10.

The effects of temperature variations on the sensitivity of the capsules and their associated linkages are typically compensated by the inclusion of bimetallic strips in the lever system. The expansion or contraction of these strips varies the degree of lever movement in the system of linkages.

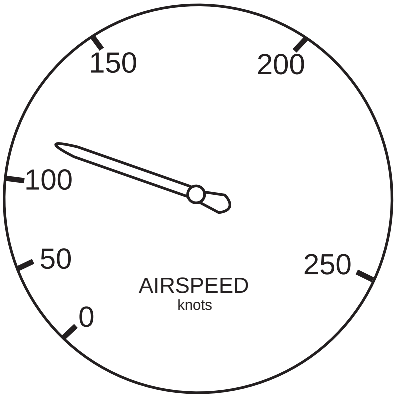

Figure 1.9 Square law calibration of ASI dial.

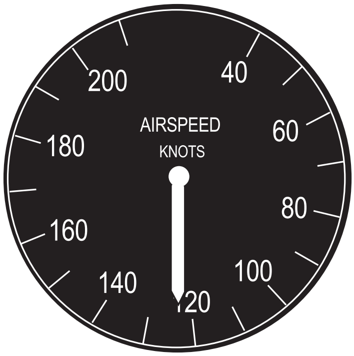

Figure 1.10 Typical ASI presentation.

The ASI scale is often marked with coloured arcs and radial lines. The arcs indicate operating speed ranges and the radial lines indicate limiting speeds. These speed ranges and limiting speeds are as follows:

V_{ne} This is the 'never exceed' speed, beyond which structural damage may occur. It is indicated on the ASI presentation by a red radial line.

V_{no} This is the maximum speed under normal operating conditions with the aircraft 'clean' (i.e. flaps and landing gear retracted).

V_{s1} This is the stalling speed with the aircraft 'clean'. A green arc extends around the ASI scale from V_{s1} to V_{no}.

V_{so} This is the stalling speed with flaps and landing gear fully extended.

V_{fe} This is the maximum permitted speed with flaps extended. A white arc extends around the ASI scale from V_{so} to V_{fe}.

V_{mc} This is the minimum control speed with one engine inoperative on a multi-engine aircraft. It is indicated by a red radial line on the ASI scale.

V_{yse} This is the best rate-of-climb speed with one engine inoperative on a multi-engine aircraft. It is indicated by a blue radial line on the ASI scale.

V_{mo}/M_{mo} Some ASIs incorporate a red and white striped pointer showing the maximum allowable airspeed (V_{mo}) which is often the mach-limiting airspeed, at which the airflow over parts of the airframe will be approaching the critical mach number (M_{crit}) for the aircraft. The pointer is actuated by a static pressure capsule and specially calibrated mechanism. This allows the V_{mo} pointer reading to increase progressively up to about 25 000 ft, as the air becomes less dense. Above this altitude the V_{mo} pointer reading is progressively decreased, since M_{crit} will be reached at progressively lower values of indicated airspeed.

V_{lo} This is the maximum speed at which the landing gear may be safely extended or retracted.

V_{le} This is the maximum speed at which the aircraft may be flown with landing gear extended.

Note: Mach number and critical mach number will be explained in detail in the section dealing with the mach meter.

ASI errors

(1) **Instrument error**. This is effectively the same as for the pressure altimeter. It is the error between the airspeed that the ASI should indicate and that which it actually does, due to manufacturing tolerances and friction within the instrument.

(2) **Position or pressure error**. This is the error caused by pressure fluctuations at the static source. These may be due to the position of the static source, hence the term position error. The error can be determined over the speed range of the aircraft and recorded on a correction card. Pressure error may also occur when the aircraft is at an unusual attitude or when flaps or landing gear are extended, in which case it is known as manoeuvring error. An example of ASI values with a normal static source in use, and the corrections to be applied when the alternate source is in use, is shown in Table 1.2 below. These data are normally found in the AOM.

(3) **Compressibility error**. At true airspeeds above about 300 knots the air brought to rest in the pitot tube is compressed to a pressure greater than dynamic pressure, causing the ASI to overread. The effect increases with altitude, since less dense air is more readily compressed. As pre-

Table 1.2 Airspeed calibration card.

Condition	Indicated airspeed (knots)										
Flaps up											
Normal	40	50	60	70	80	90	100	110	120	130	140
Alternate	39	51	61	71	82	91	101	111	121	131	141
Flaps 10°											
Normal	40	50	60	70	80	90	100	110			
Alternate	40	51	61	71	81	90	100	110			
Flaps 40°											
Normal	40	50	60	70	80	85					
Alternate	38	50	60	70	79	83					

viously stated, the compressibility error correction can be found using the circular slide rule (navigation computer).

(4) **Blocked pitot tube**. A blocked pitot tube will mean that the pressure in the pitot system will be unaffected by changes in airspeed. In level flight the ASI will indicate a false airspeed, dependent upon the pressure locked in the system, and will not indicate changes in airspeed. In the climb the decreasing static pressure will cause the ASI capsule to expand and the instrument will falsely indicate an increasing airspeed. In the descent the reverse will be the case.

(5) **Blocked static source**. A blocked static source will cause the ASI to underread as the aircraft climbs above the altitude at which the blockage occurred, since the pressure trapped in the instrument case will be progressively greater than ambient static pressure. In the descent the reverse will happen and a hazardous condition arises, since the progressively overreading ASI may cause the pilot to reduce power and airspeed may fall below stalling speed. A blockage to the static source during level flight will not be readily apparent, since the ASI indication will not change.

ASI tolerance

Typical required accuracy for the airspeed indicator is ± 3 knots.

The mach meter

The speed at which sound travels through the air is known as the speed of sound, or sonic speed. This speed varies with air temperature and therefore with location. The speed of sound at any specific location is known as the local speed of sound (LSS).

Aircraft that are not designed to fly at supersonic speeds usually suf-

fer both control and structural problems when the airflow over the airframe, particularly over the wings, approaches the LSS. Consequently it is essential that pilots of such aircraft be aware of the aircraft's speed relative to the LSS. This is especially important at high altitude, since the speed of sound decreases with temperature, and air temperature decreases with altitude in a normal atmosphere. Hence, the greater the altitude, the lower the LSS.

The aircraft's speed relative to the LSS is measured against a scale in which the LSS is assigned a value of 1. The limiting speed above which control problems may be encountered is known as the critical mach speed and is assigned a value known as the critical mach number (M_{crit}). Supposing that, for a particular aircraft, this speed happens to be 70% of the LSS, then the critical mach number would be 0.7.

The mach number (M) is the ratio of the aircraft's true airspeed (TAS) to the LSS. This may be represented by the equation:

$$\mathrm{M} = \frac{\mathrm{TAS}}{\mathrm{LSS}}$$

Thus, if an aircraft is flying at a TAS of 385 knots at an altitude where the LSS is 550 knots, the aircraft's mach number is $385 \div 550 = 0.7$. Clearly, if the aircraft were flying at a TAS of 550 knots in these conditions its mach number would be 1.0 and it would be flying at sonic speed.

Since mach number is the ratio of TAS to LSS it follows that it is proportional to the ratio of EAS, CAS or RAS to LSS, but the requisite corrections must be applied. Once again, the navigation computer, or circular slide rule, is equipped to do this.

The speed of sound in air is entirely dependent upon the air temperature. The lower the air temperature, the lower the LSS, therefore LSS decreases with increasing altitude in a normal atmosphere. The LSS can be calculated using the formula:

$$\mathrm{LSS} = 38.94\sqrt{\mathrm{TK}}$$

where TK is the local air temperature in degrees kelvin.

The kelvin, or absolute temperature, scale is based upon absolute zero, which is equal to $-273°$C. Thus, 0°C is equal to 273K and an ISA mean sea level temperature of +15°C is equivalent to $273 + 15 = 288$K. From this it can be shown that the local speed of sound at ISA mean sea level is:

$$38.94\sqrt{288} = 661\,\text{knots}$$

However, at 36 090 ft in the standard atmosphere, where the ambient air temperature is $-56.5°$C:

$$\begin{aligned} \text{LSS} &= 38.94\sqrt{273 - 56.5} \\ &= 38.94\sqrt{216.5} \\ &= 573\,\text{knots} \end{aligned}$$

Since mach number is the ratio of TAS to LSS it follows that mach number is also dependent upon local air temperature. At any given true airspeed the aircraft's mach number varies inversely with temperature. Thus, an aircraft climbing in standard atmospheric conditions at a true airspeed of 573 knots would reach mach 1 at an altitude of 36 090 feet, whereas at sea level its mach number would have been $573 \div 661 = 0.87$.

The mach meter is essentially a pressure altimeter and an ASI combined in one instrument. Its purpose is to indicate the aircraft's airspeed relative to the LSS.

We have seen that the LSS decreases with air temperature and, therefore, with altitude in a normal atmosphere. The mach meter contains an altitude capsule similar to that in the pressure altimeter. Static pressure is supplied to the sealed instrument casing and the aneroid altitude capsule will expand against a light spring as static pressure decreases with increasing altitude.

Figure 1.11 is a schematic diagram showing the principle of operation of the mach meter. The instrument also contains an airspeed capsule, the inside of which is connected to pitot pressure. As airspeed, and therefore dynamic pressure, increases the airspeed capsule will expand, exactly as in the ASI.

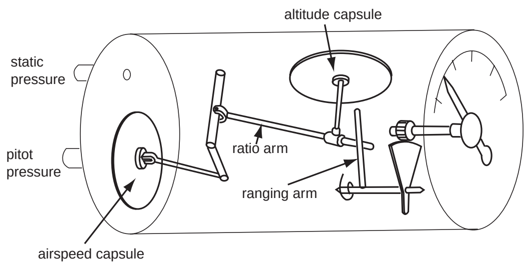

Figure 1.11 Mach meter principle of operation.

The movement of the altitude and airspeed capsules is transmitted to the mach meter pointer through a system of mechanical links and gears. The pointer rotates against a dial calibrated to show the aircraft speed in terms of mach number.

As a pressure instrument the mach meter cannot measure the ratio of TAS to LSS, but it satisfies the requirement by measuring the ratio of dynamic pressure (pitot − static) to static pressure. This can be expressed as:

$$M = \frac{(p - s)}{s}$$

From the foregoing it follows that an increase in airspeed will raise the mach number, bringing the aircraft's speed closer to the LSS. An increase in altitude will reduce the LSS, also bringing the aircraft's speed closer to the LSS, raising the mach number. Let us examine the operation of the mach meter under these circumstances.

Increased airspeed at constant altitude

Movement of the airspeed capsule is transmitted to the instrument pointer through a ratio arm, a ranging arm and gearing. Let us assume there is an increase of airspeed in level flight. The consequent increase in dynamic pressure causes the airspeed capsule to expand and the ratio arm rotates to bear against the ranging arm. This in turn rotates a quadrant and pinion gear system, which is connected to the instrument pointer. Thus, the expansion of the airspeed capsule causes the pointer to move against a calibrated scale, indicating an increased mach number.

A decrease in airspeed will cause the airspeed capsule to contract and the above sequence will be reversed, resulting in a decreased mach number indication.

Increased altitude at constant airspeed

An increase in altitude will cause the altitude capsule to expand and the linear movement of the ratio arm against the ranging arm will again rotate the ranging arm and its associated quadrant and pinion to rotate the mach meter pointer, indicating an increased mach number.

A decrease in altitude will cause the altitude capsule to contract and the above sequence will be reversed, resulting in a decreased mach number indication.

Figure 1.12 shows a typical mach meter display.

Mach/TAS calculations

Calculation of mach number given TAS and LSS has already been demonstrated. Clearly, by transposition of the formula, it is possible to calculate TAS given mach number (M) and LSS.

$$\text{TAS} = \text{M} \times \text{LSS}$$

LSS can be calculated if the ambient air temperature is known and converted to kelvin.

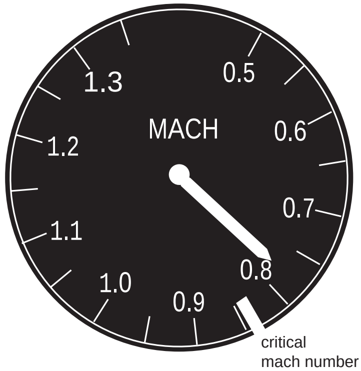

Figure 1.12 Typical mach meter display.

Mach meter errors

(1) **Instrument error**. In common with all pressure instruments the manufacturing tolerances inevitably lead to small errors of measurement due to friction and lost motion in the linkages and gearing. These are typically of the order of ± 0.1 M over a range of 0.5 M to 1.0 M. Instrument error increases slightly with increasing speed and altitude.

(2) **Pressure error**. The mach meter is more sensitive than the ASI to errors in static pressure measurement, since it uses the ratio of dynamic pressure $(p - s)$ to static pressure, whereas the ASI uses the ratio of pitot pressure to static pressure.

(3) **Blockages**. Blockage of either pressure source will cause the measured ratio to be incorrect. If the static source is blocked, changes in altitude will not be sensed and the instrument will underread in the climb. If the pitot source becomes blocked the instrument will not respond to speed changes. The exact effect of a pitot blockage depends largely upon what the aircraft is doing at the time and is therefore difficult to predict.

The vertical speed indicator (VSI)

The vertical speed of an aircraft is otherwise known as its rate of climb or descent and the VSI is alternatively known as the rate of climb/descent indicator (RCDI). The purpose of the VSI is to indicate to the pilot the aircraft's rate of climb or descent, typically in feet per minute.

Since it is *rate* of change of height being indicated it is necessary to create a pressure difference within the instrument whilst a height change is occurring, and to arrange that the magnitude of the pressure difference is proportional to the rate of change of height. This is achieved by a metering unit within the instrument, which is illustrated in schematic form in Figure 1.13.

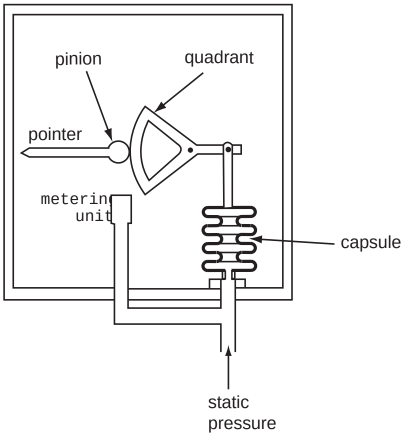

Figure 1.13 Vertical speed indicator principle of operation.

Static pressure is led directly to the inside of a capsule and also to the inside of the sealed instrument casing via a metering orifice. The capsule is connected to a pointer through linkages and a quadrant and pinion gear.

Whilst the aircraft is in level flight the pressure in the capsule is the same as that in the casing and the pointer is in the horizontal, nine o'clock position indicating zero. When the aircraft enters a climb, static pressure begins to fall and this is sensed virtually immediately within the capsule. Pressure in the instrument casing is now greater than that in the capsule, since air can only escape at a controlled rate through the restricted orifice of the metering unit. The pressure difference causes the capsule to contract, driving the pointer in a clockwise direction to indicate a rate of climb. The faster the aircraft's rate of climb, the greater will be the pressure difference between capsule and casing and the greater the capsule compression, driving the pointer further around the scale.

During a descent the increasing static pressure will be felt virtually instantly within the capsule, but the pressure in the instrument casing will rise at a slower rate due to the effect of the metering unit. Hence the capsule will expand, against a spring, driving the pointer in an anti-clockwise direction to indicate a rate of descent proportional to the pressure difference, which is in turn proportional to the rate of descent.

When the aircraft levels out at a new altitude the pressure in the instrument casing will equalise with that in the capsule and the pointer will return to zero. A typical VSI presentation is shown in Figure 1.14. It should be noted that the scale graduation is logarithmic, having greater spacing at lower rates of climb. This is deliberate, to facilitate easy interpretation when small changes of height are being made.

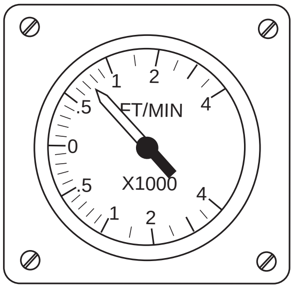

Figure 1.14 VSI presentation.

Instantaneous vertical speed indicator (IVSI)

A disadvantage of the basic VSI described above is that there is an inherent time lag in displaying climb or descent when either is initiated. This is due to the hysteresis of the capsule, which needs a significant pressure differential before it begins to expand or contract. In aircraft of moderate performance this is unimportant, but in higher performance aircraft an instantaneous response to height change is necessary.

This is achieved by introducing a small cylinder connected to the static pressure supply to the capsule, containing a lightly spring-loaded free piston. This device is known as a dashpot accelerometer. When a climb is initiated, inertia causes the piston to move down in the cylinder, creating an instantaneous, but temporary, pressure drop in the capsule. The capsule immediately contracts, causing the instrument to indicate immediately the commencement of a climb. Thereafter, the decreasing static pressure due to the climb causes an indicated rate of climb as previously described.

When a descent is initiated, inertia causes the dashpot piston to rise in its cylinder, creating a small instantaneous pressure rise in the capsule, sufficient to expand it and indicate the commencement of descent.

A minor disadvantage of the IVSI is that, on entering a turn in level flight, the centrifugal acceleration force is liable to displace the dashpot piston and cause a temporary false indication of climb. Upon exiting the turn the reverse will be the case. The IVSI is also more sensitive to turbulence and is liable to give a fluctuating indication in such conditions. This is usually damped out by the inclusion of a restriction in the static connection to the capsule and the metering unit.

An adjusting screw is provided below the face of the instrument, by which the pointer can be set to read zero. The range of adjustment available is, typically, between ± 200 ft per minute and ± 400 ft per minute, depending upon the scale range of the instrument.

Errors

(1) **Instrument error**. Manufacturing tolerances lead to small errors in the internal mechanism. When these cause displacement of the pointer from the zero position with the aircraft stationary on the ground it can be corrected with an adjusting screw on the front of the instrument.
(2) **Pressure error**. Disturbances at the static source may cause the instrument to display an incorrect rate of change of height.
(3) **Lag**. A delay of a few seconds before a rate of change of height is indicated is normal with the basic VSI. The IVSI virtually eliminates lag.
(4) **Transonic jump**. A transonic shock wave passing over the static source will cause the VSI briefly to give a false indication.
(5) **Blockage**. Blockage of the static source will render the VSI useless, since it will permanently give a zero indication.

Failure of the instrument will usually result in a fixed indication of zero rate of change of height.

Dynamic vane-type VSI

Sailplanes and a few very light aircraft often use a different type of VSI known as a variometer. Figure 1.15 shows a schematic illustration.

An enclosed casing shaped rather like a shallow tin can contains a pivoted vane, held in a central position by a light leaf spring. The casing on one side of the vane is connected to the static pressure source and on the other side of the vane it is connected to an enclosed air reservoir. In level flight air can leak past the vane sufficiently for the pressure in the reservoir to equalise with

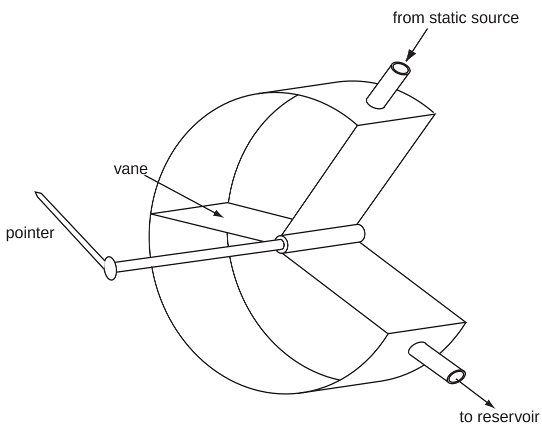

Figure 1.15 Variometer.

static air pressure. Under these conditions the leaf spring will centralise the vane, which is connected to a pointer indicating zero rate of climb/descent.

When a climb is initiated the static pressure will begin to fall below the pressure stored in the air reservoir. The differential pressure across the vane will deflect it and the attached pointer will indicate a rate of climb. The faster the rate of climb, the greater the differential pressure and the more the vane/pointer will be deflected. When the aircraft levels off, air will leak past the vane to equalise the reservoir with static pressure once more and the leaf spring will centralise the vane/pointer to the zero position. In a descent the sequence of events will be the opposite of that described above.

The vane-type variometer is incapable of displaying large rates of height change and is therefore unsuitable for most powered aircraft. It does, however, have the advantage that it suffers less from time lag (i.e. initial response) than the basic VSI.

There is also a type of variometer used specifically in sailplanes that responds to height gain or loss not initiated by the pilot, as in a thermal. This type is known as the total energy variometer.

The air data computer (ADC)

In addition to the four instruments already covered there are numerous systems in modern aircraft that require air data inputs in terms of static pressure, pitot pressure and air temperature. Since most, if not all, of these systems are electronic in operation it is logical to supply such data in electronic form. The air data computer receives pitot and static pressure from the normal and alternate sources and converts these into electrical signals and transmits them to the various indicators and systems. As far as indicators are concerned, this removes the need for bulky instruments that take up significant panel space and means that the information can be presented in digital form if necessary.

The centralised air data computer can also be programmed to apply the necessary corrections for pressure error, barometric pressure changes and compressibility effects. With the addition of air temperature data inputs, true airspeed can be calculated by the computer.

Analogue and digital ADCs

Air data computers are usually of the digital type; that is, they transmit data in digital format which is compatible with other computer-based systems. Analogue air data computers, which transmit their output data to servo-operated devices, are less common, although a few are still in existence.

Figure 1.16 is a block diagram showing the data inputs and outputs of a typical ADC. It will be seen that the data inputs are pitot and static pressure

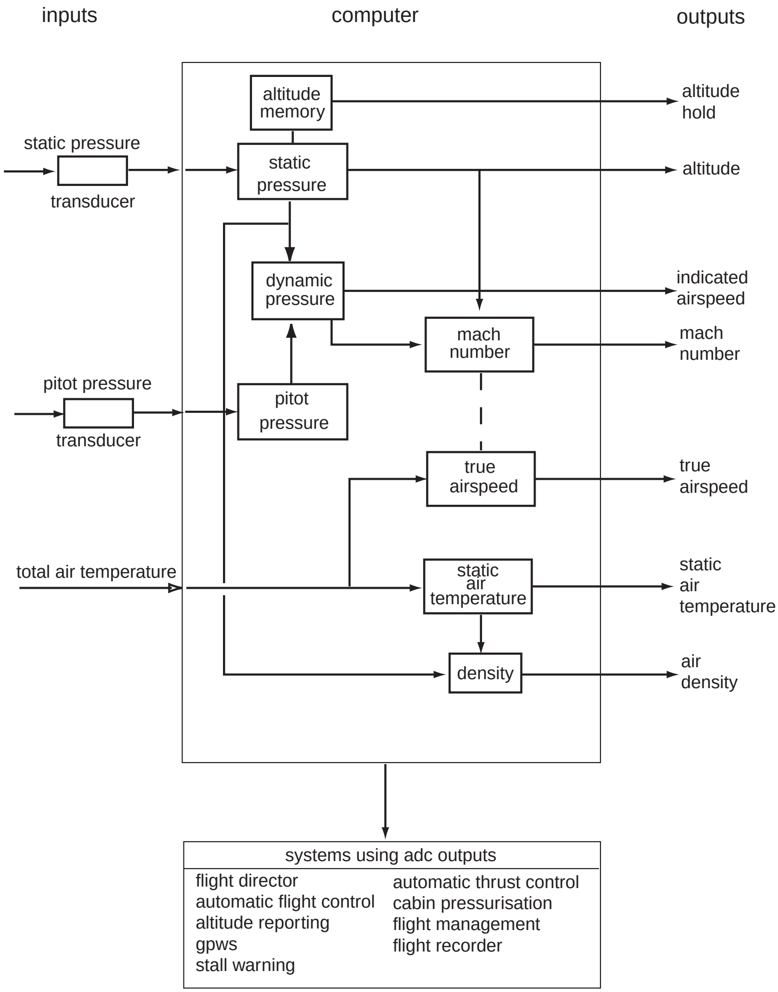

Figure 1.16 Data inputs and outputs of an air data computer.

and total air temperature (TAT). From these, electrical signals are generated and transmitted as electronic data to operate the pilots' air data instrument displays, plus TAS, TAT and SAT (static air temperature) displays. Additionally, these signals are transmitted to the various flight management control systems, listed in Figure 1.16. Loss of air data input activates a warning logic circuit within the ADC, which causes warning flags to appear on the associated indicators and annunciators to illuminate on the computer control panel.

At this point it is perhaps appropriate to define the various forms of air temperature measurement.

(1) **SAT (static air temperature)**. This is the temperature the air at the surface of the aircraft would be at if there were no compression effects due to the aircraft's movement. At very low airspeeds these effects are negligible, but for most transport aircraft normal flight speeds are such that the direct measurement of SAT is virtually impossible. SAT is also known as outside air temperature (OAT).

(2) **TAT (total air temperature)**. Total air temperature is the temperature of the air when it has been brought completely to rest, as in the pitot tube. The ram rise, that is the temperature increase due to compression, can then be subtracted from TAT to give corrected outside air temperature (COAT). The value of ram rise can be calculated for any given mach number, so clearly the air data computer can be programmed to make this correction.

Temperature measurement probes

Aircraft that operate at low airspeeds, such as some helicopters and light aircraft, usually employ a simple bimetallic thermometer, which operates a rotary pointer against a temperature scale to indicate what is essentially static air temperature.

TAT sensors are more complex because they must, as far as possible, recover the temperature rise due to adiabatic compression of the air as it is brought to rest; what is known as the 'ram rise'. To do this it is necessary to use sophisticated probes to capture and slow the air and then convert the air temperature to an electrical signal for transmission to the ADC.

The sensitivity of the probe in terms of the extent to which it truly achieves, or *recovers*, the full ram rise effect is known as its recovery factor. For example, a probe that senses SAT plus 85% of the ram rise in temperature would be said to have a recovery factor of 0.85.

TAT probes are typically contained within an aerodynamically shaped strut with an air intake mounted on the outer end, clear of any boundary layer air. Air is drawn into the hollow strut, through the air intake, where it is brought virtually to rest. Its temperature is sensed by a platinum resistance-type element mounted within the strut, which produces an electrical signal proportional to the temperature. Most modern TAT probes have a very high recovery factor, usually very close to unity (1.0).

An alternative type of probe uses engine bleed air to create a reduction of pressure within the casing of the probe. This has the effect of drawing air into the hollow strut at a higher rate, so that the de-icing heating element within the strut cannot affect the sensed temperature of the indrawn air.

Figure 1.17 is a diagram of an air temperature measurement probe.

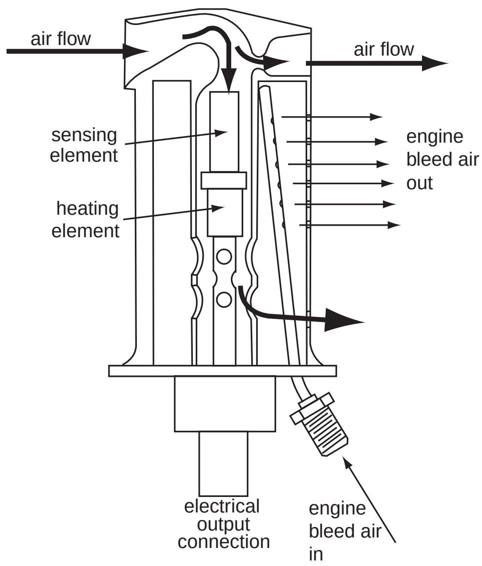

Figure 1.17 TAT measuring probe.

Pressure transducers

Devices called transducers convert pitot and static pressure into suitable electrical signals for transmission to the air data computer. In the case of analogue ADCs the transducer is often of the electro-magnetic type, the amplified output of which drives a servomotor and operates a synchro system, which in turn operates the analogue instrument displays.

Digital ADCs more commonly utilise piezoelectric transducers that form part of a solid-state circuit. Some crystalline materials, such as quartz, can be made to generate varying electrical signals when subjected to pressure. A diaphragm composed of thin quartz discs impregnated with metallic particles is subjected to pitot or static pressure and the subsequent flexing of the diaphragm creates an electrical charge in the discs, the polarity of which is dependent upon the direction of flexing. Thus a signal proportional to increasing or decreasing pressure is generated.

Sample questions

1. At lower altitudes, near to sea level, the change of atmospheric pressure with height is approximately:

   a. 1 hPa per 50 ft?
   b. 1 hPa per 27 ft?
   c. 27 hPa per 1 ft?
   d. 1 hPa per 10 ft?

2. ISA mean sea level pressure is:

   a. 1013 hPa?
   b. 1025 hPa?
   c. 29.02 in Hg?
   d. 1030 hPa?

3. An aircraft is at 600 ft above an airfield, the elevation of which is 250 ft amsl. With QFE set, the altimeter will read:

   a. 850 ft?
   b. 350 ft?
   c. 600 ft?
   d. 250 ft?

4. An aircraft altimeter has been set for a regional QNH of 1010 hPa, but it is operating in an area where msl pressure is 1000 hPa. The altimeter is indicating 2000 ft; actual height amsl is:

   a. 2300 ft?
   b. 1730 ft?
   c. 1940 ft?
   d. 1970 ft?

5. Which of the following is correct?

   a. Static pressure = pitot pressure + dynamic pressure?
   b. Dynamic pressure = pitot pressure + static pressure?
   c. Pitot pressure = dynamic pressure − static pressure?
   d. Dynamic pressure = pitot pressure − static pressure?

6. An aircraft is flying at constant IAS:

   a. As altitude decreases EAS will increase?
   b. As altitude increases CAS will increase?
   c. As altitude increases TAS will increase?
   d. As altitude decreases TAS will increase?

7. The stalling speed with flaps and landing gear fully extended is assigned the notation:

   a. V_{so}?
   b. V_{se}?
   c. V_{fe}
   d. V_{no}?

8. An aircraft is flying at a TAS of 400 knots at an altitude where the LSS is 550 knots. The aircraft's mach number is:

   a. 1.4?
   b. 0.65?
   c. 0.82?
   d. 0.73?

9. The white arc on an ASI scale:

   a. Extends over the safe speed range with flaps retracted?
   b. Extends over the safe speed range with flaps fully extended?
   c. Extends over the safe speed range with one engine inoperative?
   d. Extends over the speed range from V_{no} to V_{ne}?

10. The error suffered by air data instruments that is caused by manufacturing tolerances is known as:

    a. Pressure error?
    b. Lag?
    c. Position error?
    d. Instrument error?

11. The type of VSI commonly used in sailplanes is:

    a. IVSI?
    b. Simple VSI?
    c. Dynamic vane type?
    d. Variation meter?

12. The data inputs to an ADC are:

    a. TAT, pitot and static?
    b. SAT, pitot and static?
    c. COAT, dynamic and static?
    d. OAT, pitot and static?

Chapter 2
Gyroscopic Instruments and Compasses

Gyro fundamentals

Gyroscopic instruments are of great importance in aircraft navigation because of their ability to maintain a constant spatial reference and thereby provide indication of the aircraft's attitude. The principal instruments that use the properties of the gyroscope are the directional gyro, the artificial horizon or attitude indicator and the turn and bank indicator.

Gyroscopic properties

The gyroscope used in these instruments comprises a rotor, or wheel, spinning at high speed about an axis passing through its centre of mass and known as the spin axis. A simple gyro rotor is illustrated in Figure 2.1. When a rotor such as that in Figure 2.1 is rotating at high speed it exhibits two basic properties, known as rigidity and precession. It is these properties that are utilised to give gyroscopic instruments their unique features.

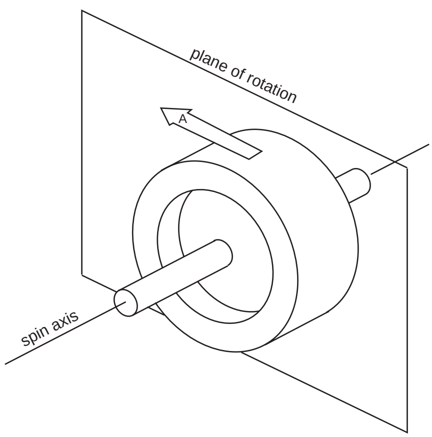

Figure 2.1 Gyro rotor.

Rigidity

The spinning rotor of the gyro has rotational velocity and therefore, if we consider any point on the rotor, that point has angular velocity as indicated by the arrow A in Figure 2.1. Since the rotor has mass, that angular velocity produces angular momentum, which is the product of angular velocity and mass. As stated in Newton's First Law of Motion, any moving body tends to continue its motion in a straight line and this is known as inertia. In the case of the spinning gyroscope there is a moment of inertia about the spin axis, which tends to maintain the plane of rotation of the gyro. Consequently, the spin axis of a gyroscope will maintain a fixed direction unless acted upon by an external force. This property is known as *rigidity*. Another way of putting this is that the spin axis of the gyro will remain pointing toward a fixed point *in space* unless it is physically forced to move.

Since rigidity is the product of angular velocity and mass it follows that the rigidity of a gyroscope may be increased by increasing either its angular velocity, or its mass, or both. Increasing the speed of rotation of the rotor, or its diameter, will increase angular velocity and therefore angular momentum. The rotor diameter is constrained by the need to keep the instrument as compact as possible and so the gyro rotor is made to spin at very high speed. Similarly, the mass of the rotor is constrained by its size limitations, but angular momentum is improved if the mass is concentrated at the rim of the rotor, as seen in Figure 2.1.

Precession

Precession is defined as the angular change in direction of the spin axis when acted upon by an applied force. Let us suppose that the axis of the gyro rotor in Figure 2.2 has a force applied to it as shown in (a). Application of a force to the spin axis as shown is exactly the same as if the force had been applied at point X on the rotor. If the rotor were stationary then it, and its spin axis, would tilt as shown at Figure 2.2(b). However, when the rotor is rotating it has not only the applied force acting upon it, but also the angular momentum previously described. The combination of the two displaces the effect of the applied force through 90° in the direction of rotation, as shown in Figure 2.2(c). Thus, the spin axis of the gyroscope will precess as shown in Figure 2.2(d) in response to the force applied in Figure 2.2(a).

The *rate* at which a gyro precesses is dependent upon the magnitude of the applied force and the rigidity of the rotor. The greater the applied force, the greater the rate of precession. However, the greater the rigidity of the rotor the slower the rate of precession for a given applied force.

A gyro will continue to precess so long as the applied force is maintained, or until the applied force is in the same plane as the gyro plane of rotation, as

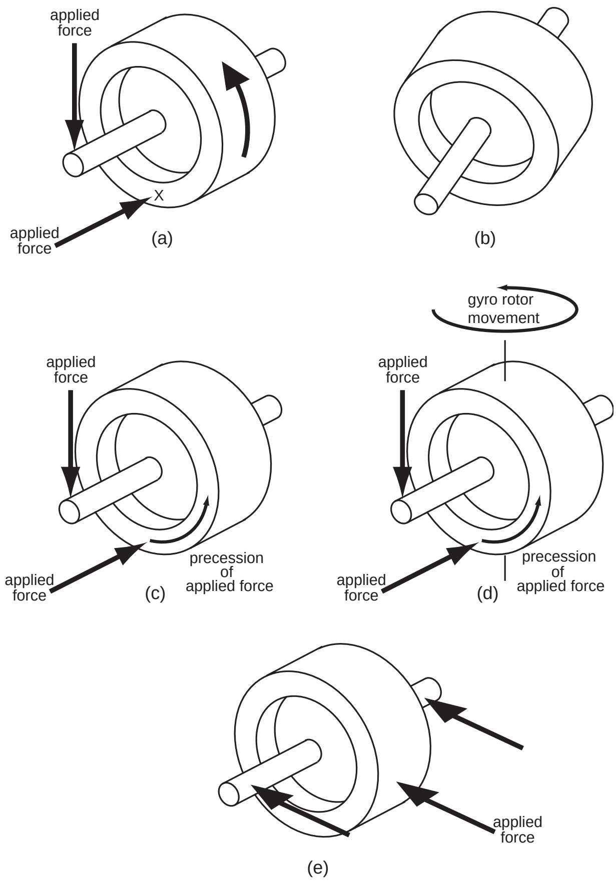

Figure 2.2 Gyroscopic precession.

shown in Figure 2.2(e). If the applied force is removed, precession will immediately cease.

Free gyro

Clearly the rotor of the gyroscope must be contained within a supporting structure. The rotor spindle is mounted within a ring known as a gimbal and this is in turn mounted within a framework, the design of which depends upon the gyro function. All gyroscopes must have freedom for the rotor to rotate and to precess. The gyroscope cannot precess about the axis of rotation, but precession may take place about either of the two axes at right angles to the plane of rotation. A gyroscope that has freedom to precess about both these axes is known as a *free gyro*, and is said two have two degrees of freedom of precession. Such a gyroscope is illustrated in Figure 2.3. The number of

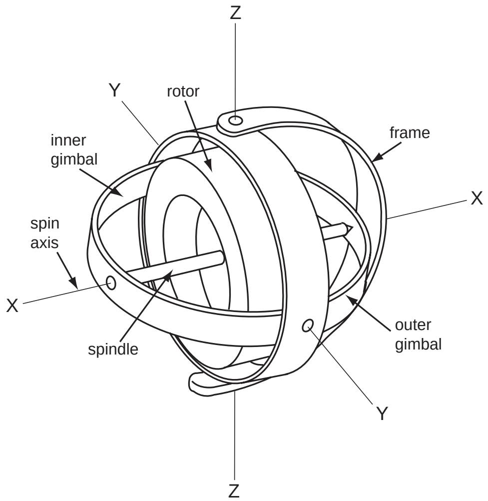

Figure 2.3 Free gyro.

degrees of freedom of precession of any gyroscope is the same as its number of gimbals. It will be seen that the gyro rotor spindle is mounted in bearings within a ring, or gimbal, known as the inner gimbal. This is in turn mounted in bearings that are attached to a second gimbal ring, known as the outer gimbal. Thus, the gyro rotor is free to spin about spin axis XX and it also has freedom of movement about the inner gimbal axis YY. The outer gimbal is mounted in bearings attached to the frame of the assembly and therefore has freedom of movement about the third axis, ZZ.

If the frame of the free gyro were to be fixed to the instrument panel of an aircraft, the aircraft could be pitched, rolled or even inverted and the spin axis of the spinning gyroscope would remain aligned with the same fixed point in space. In point of fact the free gyro has no practical application in aircraft, but gyroscopes having freedom of precession about two axes, known as *tied gyros*, are extremely useful.

Tied gyro

The aircraft instruments that employ gyros provide a fixed reference, about which aircraft movement is indicated to the pilot. The directional gyro provides the pilot with aircraft heading information and so its reference axis is the aircraft's vertical axis and the gyro rotor must be sensitive to movement about that axis and no other. The function of the attitude indicator is to provide the pilot with indications of aircraft attitude with reference to the pitch and roll axes of the aircraft and so its gyro must be sensitive to aircraft movement about these axes. The turn indicator is

required to indicate rate of turn and so it must be sensitive to aircraft movement about the vertical axis.

A gyroscope is not sensitive to movement about its spin axis, so its rotor must be maintained at right angles to the required axis for maximum sensitivity. Consider the situations depicted in the illustration in Figure 2.4.

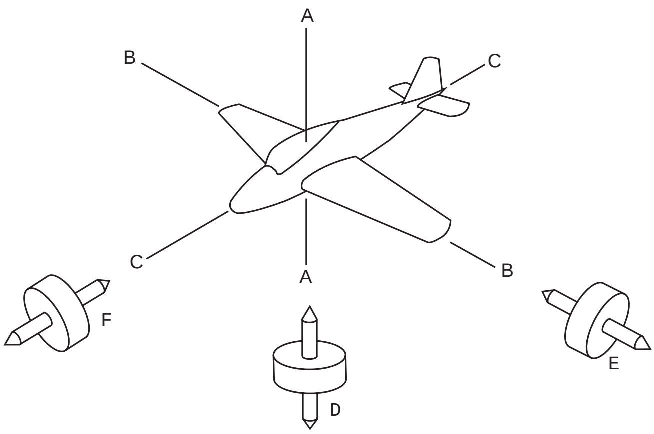

Figure 2.4 Gyroscope spin axis alignment.

Axis AA is the aircraft's vertical, or yaw, axis. Any movement about this axis involves a change in aircraft heading and so we require the directional gyro to be sensitive to movement about this axis. Since a gyro is not sensitive to movement about its spin axis it is clear that a gyro with its spin axis vertical (gyro D in Figure 2.4) would not be suitable, but that the spin axis of a directional gyro must be maintained horizontal.

The attitude indicator is required to indicate aircraft attitude with reference to the aircraft pitch and roll axes, BB and CC respectively in Figure 2.4. Clearly its spin axis must not be aligned with either of these aircraft axes. Consequently gyro E would not be suitable, because its spin axis is aligned with the aircraft's pitch axis, and gyro F would not be suitable, because its spin axis is aligned with the aircraft's roll axis. Thus, the spin axis of the attitude indicator gyro must be maintained vertical as in gyro D, and not just aircraft vertical, but *earth* vertical.

The turn indicator is required to indicate *rate* of turn, that is the rate at which the aircraft is turning about its vertical axis. The turn indicator gyro must therefore be sensitive to movement about the *aircraft* vertical axis AA and so its spin axis must be aligned with either axis BB or CC. For practical reasons that will become apparent when we study this instrument in detail, it is aligned with the aircraft's lateral axis BB, as for gyro E. The gyro of the turn indicator is known as a rate gyro.

In each case we require the spin axis of the gyroscope to be *tied* to a particular direction, e.g. earth vertical or aircraft horizontal. Such a gyroscope is known as a tied gyro. A tied gyro that is controlled by the earth's gravity is also known as an earth gyro; this is the case with the attitude indicator.

Another type of gyro, which we will meet later when we study the gyro-stabilised platform of an inertial navigation system, is the rate integrating gyro. This is a single degree of freedom gyro, sensing rate of movement about one axis only, which is integrated to give change of distance.

Gyroscope drift (wander) and topple

Earlier in this dialogue it was stated that the spin axis of a gyroscope remains aligned with some point in space, as opposed to alignment with any earth reference such as true or magnetic north. Any deviation of a horizontally aligned spin axis from its point of reference is known as gyro drift, or wander. Gyro drift is of two types, real drift and apparent drift. Deviation of a vertically aligned spin axis from its reference is known as gyro topple.

Real drift

As we have seen, the gyro comprises a spinning rotor mounted in a gimbal, which is in turn pivoted to either another gimbal or a frame. If the rotor, its spindle or a gimbal is not perfectly balanced the imbalance will apply a force to the rotor. This force will cause precession, which will cause the spin axis of the gyro to deviate from its spatial reference. The same effect can arise due to friction or wear in the rotor spindle bearings. The drift due to the spatial deviation is known as real or random drift; it is usually very small and it cannot be calculated, so it is impossible to produce correction charts for real drift.

Apparent drift

Let us now consider the case of a gyroscope with its spin axis tied to horizontal, as in the case of the directional gyro. Imagine this gyro is at the true north pole, where all directions are south. The spin axis of the gyroscope has been aligned with the Greenwich meridian, 0° of longitude. Now remember that a gyroscope alignment is really with some point in space and it is to this unknown point that the spin axis is truly pointing. The earth rotates at 15° per hour and so, assuming that our gyro is perfect and does not suffer from any real drift, after one hour its spin axis will still be aligned with the same point in space. However, to the earthbound observer it will no longer be aligned with 0° of longitude, but will appear to have drifted by 15°. This is known as apparent drift due to earth rotation.

If the same gyro were to be taken to any point on the equator and aligned with true north it would not suffer at all from apparent drift due to earth rotation, because the earth reference point and the space reference point are in alignment and remain so. Thus, the rate of apparent drift due to earth rotation varies with latitude and can be calculated, since it varies as the sine of the latitude. Apparent drift due to earth rotation is given as 15 $\times$ sin latitude°/hr. Apparent drift is illustrated in Figure 2.5.

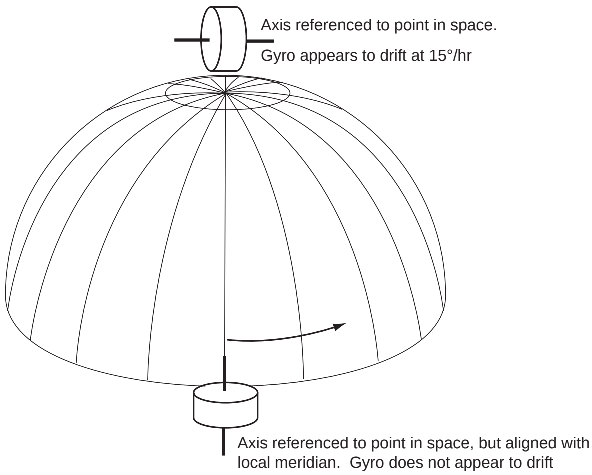

Figure 2.5 Apparent drift due to earth rotation.

Transport drift

Since the earth rotates about its north–south axis at 15° per hour it follows that an aircraft flying on a westerly heading, against the direction of earth rotation, will experience a greater rate of longitude change and one flying on an easterly heading a lesser rate. To an observer on the aircraft, the spin axis of a horizontally aligned gyro would appear to drift even though no change of latitude has occurred. This apparent drift is known as transport drift.

Apparent topple

A vertical axis gyro will also suffer apparent wander, which is conventionally known as topple. Suppose a vertical axis gyro is taken to the true north pole. Its spin axis will be aligned with the earth's spin axis and pointing toward some point in space. The earth reference and the space reference will remain in alignment and there will be no gyro topple. Suppose now the same gyro were to be taken to a point on the equator and started spinning with the spin axis perpendicular to the earth's surface, i.e. earth

vertical. After one hour the earth will have rotated 15°. The gyro spin axis will have maintained its spatial alignment and will appear to have toppled by 15°. As with apparent drift, the rate of topple is dependent upon the latitude at which the gyro is located, but in this case it varies as the cosine of the latitude. Apparent topple is given as 15 × cos latitude°/hr. Apparent topple is illustrated in Figure 2.6.

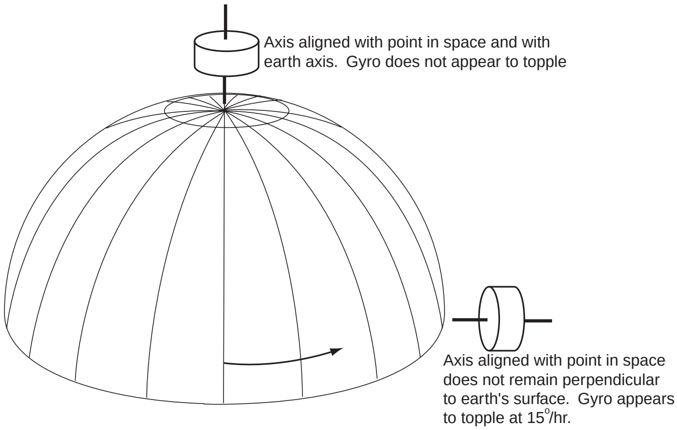

Figure 2.6 Apparent topple due to earth rotation.

Ring laser gyro

Unlike the conventional gyroscopes described above, the ring laser gyro is a solid state device that does not have any moving parts. A simplified diagram of a ring laser gyro is shown in Figure 2.7.

The device is made from a block of very expensive glass, within which there is a triangular cavity of exact dimensions, filled with a suitable lasing medium, such as helium–neon. Each side of the triangular cavity is exactly

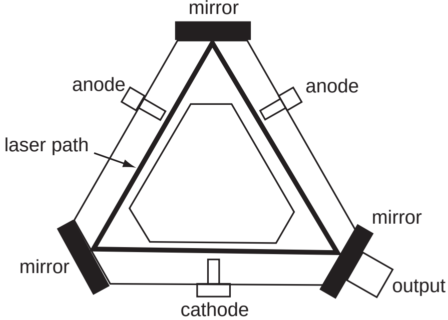

Figure 2.7 Ring laser gyro.

the same length and at each of the three junctions is a mirror, one of which is partially transmitting. At the mid-point of one side of the triangular cavity is a cathode and in the other two sides an anode is positioned at exactly the same distance from the mirrors. Laser beams travelling between the cathode and each anode will take exactly the same length of time to travel exactly the same distance.

However, if the ring laser gyro is rotated about the axis perpendicular to the laser path, one laser beam will arrive at one anode slightly before the other beam arrives at the other anode, and the time difference will be proportional to the rate of rotation. The direction of rotation will determine which laser has the shorter distance to travel. The time difference is measured and used to produce a digital readout of rate and direction of rotation.

Ring laser gyros, although very expensive to produce, have the advantage of being much more reliable than conventional gyros, because there are no moving parts subject to wear. Also they are available for immediate use when switched on, whereas conventional gyros take some time to spin up and stabilise.

Gyro drives

The rotors of gyroscopic instruments must spin at high speed to give the degree of rigidity needed and the motive power for them is either pneumatic or electrical.

(1) **Pneumatic drive**. Air-driven gyro rotors are typically powered from the aircraft's vacuum system, air being drawn through the instrument by an engine-driven vacuum pump that maintains approximately 4 in Hg of vacuum in the system. A schematic diagram of a typical light aircraft vacuum system is shown in Figure 2.8. Pneumatically driven gyros in aircraft that operate at high altitude are usually supplied with air pressure rather than vacuum, because of the difficulty in producing the requisite vacuum in a low-pressure environment. In either case the air entering the instrument is directed onto bucket-shaped indentations in the rim of the gyro rotor, driving it as a simple turbine.

    In some light aircraft the vacuum is produced by means of a venturi tube placed in the airflow. Because the device only operates in flight at speeds in excess of 100 knots and is susceptible to icing, it is unsuitable for use in aircraft where instrument flight may be required.

(2) **Electrical drive**. Alternating current (a.c.) or direct current (d.c.) motors are also used to drive gyroscopic instrument rotors, using power from the aircraft electrical systems. As a general rule, a.c. motors are preferred for attitude indicators and d.c. motors for turn indicators. Simple

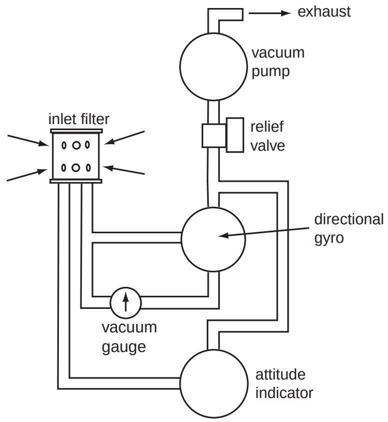

Figure 2.8 Typical light aircraft vacuum system.

direction indicators are usually air-driven, but those forming part of a magnetic heading reference system, such as the slaved gyro compass, are normally driven by electric motors. In some aircraft the main panel instruments may be electrically driven and the standby instruments air-driven.

Electrically powered gyros are necessary in aircraft intended for high altitude flight. Because they are capable of much higher rotational speeds than pneumatically powered instruments they offer increased stability and lighter construction.

In either case it is clear that interruption of the power supply will render the gyros unserviceable and it is important that the pilot should be immediately aware of this. Loss of power, whether pneumatic or electrical, will be indicated by a warning flag on the face of the instruments.

Directional gyro (DG)

The function of the directional gyro is to indicate the aircraft heading, utilising the rigidity of a spinning gyroscope so to do. The gyro spin axis is maintained horizontal and it can be set so that it is referenced to either magnetic north or true north. It will then hold this reference whilst the aircraft heading changes. A compass scale is attached to the outer gimbal of the gyroscope. The instrument casing, which is of course attached to the aircraft, moves around the fixed reference scale card as the aircraft changes heading. As a two-gimbal gyro, it has two degrees of freedom of precession.

Figure 2.9 shows the operating principle of the directional gyro. The rotor of the directional gyro is mounted in an inner gimbal ring with its spin axis

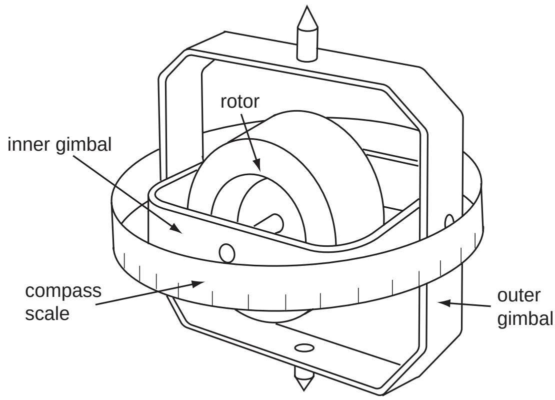

Figure 2.9 Directional gyro principle of operation.

horizontal, so that it is free to rotate in the vertical plane. The inner gimbal is in turn pivoted to an outer gimbal, so that it is free to move about a horizontal axis at right angles to the spin axis. The outer gimbal is pivoted to the case of the instrument and is free to rotate about the vertical axis. The compass scale card is attached to the outer gimbal and is typically marked from 0° to 360°.

The rotor is normally driven by air drawn in through the aircraft's vacuum system and directed by a nozzle onto buckets machined in the rim of the rotor. The rotor typically rotates at about 12 000 rpm.

Adjustment procedure

Before the start of a flight the directional gyro must be set up so that the rotor is spinning with its spin axis horizontal and the heading indication agrees with the aircraft compass reading. This is achieved with a caging mechanism and an adjustment knob on the face of the instrument.

Depressing the adjustment knob engages a caging mechanism that locks the inner gimbal in a horizontal position. It also engages a pinion with a bevel gear attached to the outer gimbal of the instrument. Rotating the adjustment knob will rotate the outer gimbal and its attached compass card and this is done until the lubber line on the face of the instrument is aligned with the required heading. Once this is satisfactory the adjustment knob is pulled out to disengage the caging mechanism and bevel gear, leaving the gyro spin axis free to maintain its fixed reference. This adjustment can also be made in flight, but must be done with the aircraft flying straight and level. The instrument should also be caged during violent manoeuvres to prevent the gyro from toppling.

Erection system

During a change of heading an aircraft is turning about its vertical, or yaw, axis. Whilst it is doing so the aircraft is, of course, banked and so the spin axis of the directional gyro must also be tilted to keep it aircraft horizontal. As we know, to move the spin axis of a gyro away from its fixed reference it is necessary to precess the gyro, and this is achieved through the design of the nozzle that directs air onto the rim of the rotor.

In later designs of air-driven directional gyros the air from the rotor is exhausted onto a wedge attached to the outer gimbal, as shown in Figure 2.10. Whilst the spin axis of the rotor remains aircraft horizontal the spin axis and the outer gimbal are mutually perpendicular and the exhaust air strikes both sides of the wedge equally, as seen in Figure 2.10.

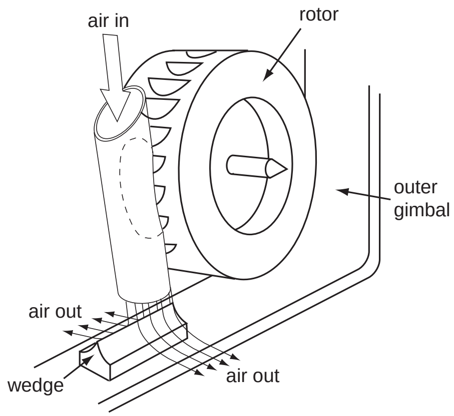

Figure 2.10 Rotor erection system – directional gyro.

When the aircraft begins to bank in a turn the outer gimbal banks with it and the rotor axis is no longer at right angles to the outer gimbal. Exhaust air now strikes one side of the wedge more than the other. This applies a force to the outer gimbal that is tending to rotate it about the vertical axis, which is the same as applying a force to one side of the gyro rotor. That force is precessed by the rotor through 90° in the direction of rotation, tilting it to keep its spin axis aircraft horizontal. Any tendency of the rotor to move from the aircraft horizontal reference will be corrected by this device.

Earlier DGs used a split, or bifurcated, air nozzle to achieve the same result.

Gimbal error

Gimbal error is when the gimbals of a gyroscope are not mutually perpendicular and the gyroscope itself is displaced. Briefly, the reason for these errors is because the spin axis of the DG is aligned with east–west on the instrument compass card. Thus, with the DG properly set, the rotor spin axis will be at right angles to the aircraft longitudinal axis (and therefore the outer gimbal) when the aircraft is on an east or west heading and aligned with it on a north or south heading. On these headings the DG will only suffer gimbal error if the aircraft is pitched and rolled simultaneously. On other headings it may occur during either pitch or roll attitude changes. The effect of gimbal error is that the instrument will give a false heading indication whilst the manoeuvre is in progress, but it will normally indicate correctly once the aircraft is returned to straight and level flight.

Drift calculations

The apparent drift due to earth rotation for a gyro that has no random (real) drift and that is stationary on the ground (i.e. not affected by transport drift) can be calculated given the latitude at which the gyro is located and the hemisphere, north or south.

In the northern hemisphere a gyro that has been aligned with north will appear to drift at the rate of $15 \times \sin$ latitude $^\circ$/hr and the DG indication will *decrease* at that rate. Reference to Figure 2.11(a) will show why this is the case. Let us assume that the aircraft in which the DG is installed is stationary on the ground at latitude 50°N, that it is on a westerly heading, and that the DG has been aligned with the local meridian and is indicating a heading of 270°. This is the situation at point A in Figure 2.11(a). The gyro drift rate will be $15 \times \sin 50$ °/hr, which is $15 \times 0.766 = 11.5$°/hr. After one hour has elapsed the gyro has not moved, but the earth has rotated and the situation will be as at point B. The gyro will appear to have drifted by 11.5° and its reading will have decreased by that amount, because its space reference is now 11.5° to the east of the local meridian. Consequently, it will now be indicating 258.5°. Without adjustment the gyro indication would continue to decrease at the rate of -11.5°/hr.

In the southern hemisphere a similar situation occurs and this is illustrated in Figure 2.11(b). With the aircraft on the ground at 50°S and the gyro indicating 270° at point C the apparent drift rate due to earth rotation will again be 11.5°/hr, but now the readings will be *increasing* at that rate. This is because, after one hour has elapsed, the gyro's space reference will lie 11.5° to the west of the local meridian, in position D, and so the DG will indicate a heading of 281.5°.

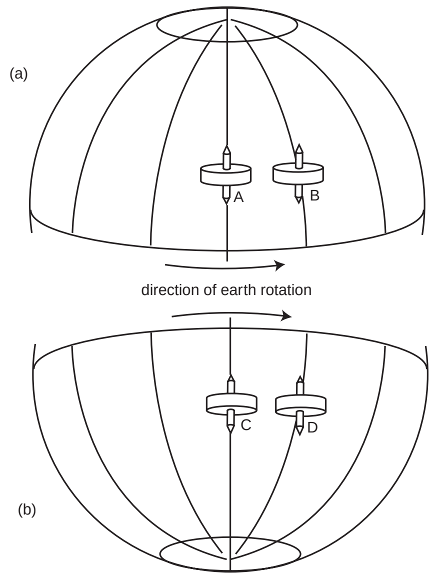

Figure 2.11 DG drift due to earth rotation.

Drift compensation

Clearly a heading indicator that was incapable of maintaining an accurate heading indication would be of no use and it will come as no surprise to learn that the directional gyro contains a compensation device. This is illustrated in Figure 2.12. Attached to the inner gimbal of the gyro is a threaded spindle with a nut attached. The gimbal is manufactured with a slight imbalance such that it is perfectly balanced with the nut, known as a latitude nut, in approximately a mid-position on its spindle. If the nut is screwed outward on the spindle it will apply a downward force to the gyro rotor, which will be precessed 90° in the direction of rotation to cause the gyroscope to precess about the vertical axis. Screwing the latitude nut inward will allow the slight imbalance of the inner gimbal to apply a force in the opposite direction.

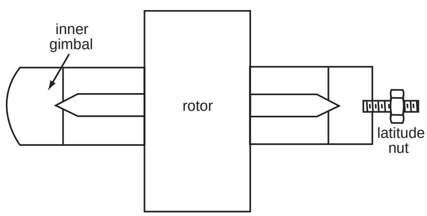

Figure 2.12 Latitude nut.

For the northern hemisphere situation described above, the latitude nut would be adjusted to precess the gyro at the rate of $+11.5^\circ/\text{hr}$, thus exactly compensating for the apparent drift rate of $-11.5^\circ/\text{hr}$. In the southern hemisphere the adjustment would be in the opposite direction.

Comparison with magnetic compass

The latitude nut of the directional gyro provides compensation only at the latitude for which it is set. This setting can only properly be made with the instrument in a workshop. If the instrument is transported north or south of the set latitude it will begin to suffer from apparent drift due to earth rotation, and the further it is moved the greater will be its error rate. Consequently, the DG must always be referenced to the aircraft magnetic compass and this can be readily done in level flight or on the ground with the adjustment knob.

An advantage of the directional gyro is that it does not suffer the turning and acceleration errors of the magnetic compass and so its heading information tends to be more accurate, especially in a steady level turn.

In the event of failure of the gyroscope the rotor will almost inevitably topple and its indication will be useless. Under these circumstances a warning flag will appear, to obscure the display.

Effect of friction

Friction or wear in the bearings of the gyro rotor will cause real, or random, drift. In a properly maintained DG this should be insignificant.

Attitude indicator (artificial horizon)

The attitude indicator uses a vertical earth gyro that has freedom of movement about all three axes. The gyro spin axis is maintained earth vertical, using the force of gravity to keep it aligned with the earth's centre. Attitude indicators may be pneumatically or electrically driven. The purpose of the instrument is to provide the pilot with an indication of the aircraft attitude in both pitch and roll.

Construction and principle of operation

Figure 2.13 illustrates the principle of operation of a pneumatic attitude indicator. The vertical gyro rotates at about 15 000 rpm and is contained within an inner gimbal. It is maintained earth vertical, thus spinning in the earth horizontal plane, by utilising gravity. The inner gimbal is pivoted to an outer gimbal with the pivot axis lying parallel to the aircraft lateral axis. The

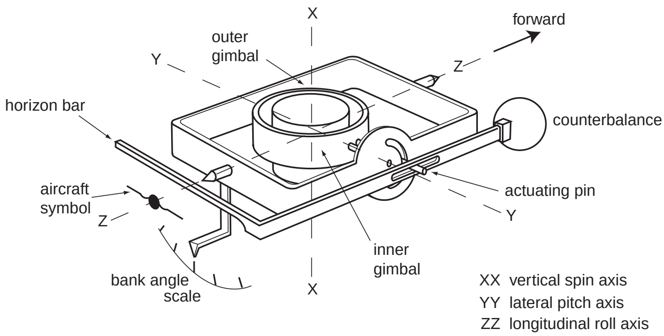

Figure 2.13 Attitude indicator – principle of operation.

outer gimbal is in turn pivoted to the instrument casing with the pivot axis lying parallel to the aircraft longitudinal axis.

Since the instrument casing is attached to the airframe it follows that any change in aircraft attitude must take place about the vertically referenced gyro. Thus, if the pitch attitude changes, the outer gimbal will pitch up or down relative to the gyro spin axis. If the roll attitude changes the outer gimbal will roll left or right relative to the gyro spin axis.

Attached to the outer gimbal is a sky plate, which is viewed through the face of the instrument. The upper half of the plate is typically coloured pale blue to represent the sky and the lower half black to represent the earth, the two divided horizontally to represent the earth's surface. Also attached to the outer gimbal by a pivoted spindle is a bar, which extends across the front of the sky plate parallel to the dividing line. This bar, known as the horizon bar, is driven by a spindle attached to the inner gimbal. Printed on, or attached to, the glass cover of the instrument is a fixed symbol representing the aircraft.

With the aircraft flying straight and level, the gyro spin axis will be perpendicular to both the lateral and the longitudinal aircraft axes and the horizon bar and aircraft symbol will appear in the mid-position, as shown in Figure 2.14(a). If the aircraft is pitched nose-up, the outer gimbal will be pitched up with it, raising the front of the gimbal relative to the gyro spin axis, which remains earth vertical.

Because the horizon bar is pivoted to the forward end of the outer gimbal, that end of the bar will rise and pivot about the actuating pin protruding from the inner gimbal. This causes the horizon bar to move down relative to the aircraft symbol, indicating a climb. In a descent the reverse happens and the horizon bar moves up relative to the aircraft symbol. These indications are represented in Figures 2.14(b) and 2.14(c).

When the aircraft is rolled to left or right the bank indication is given by the position of the horizon bar relative to the aircraft symbol. This is because

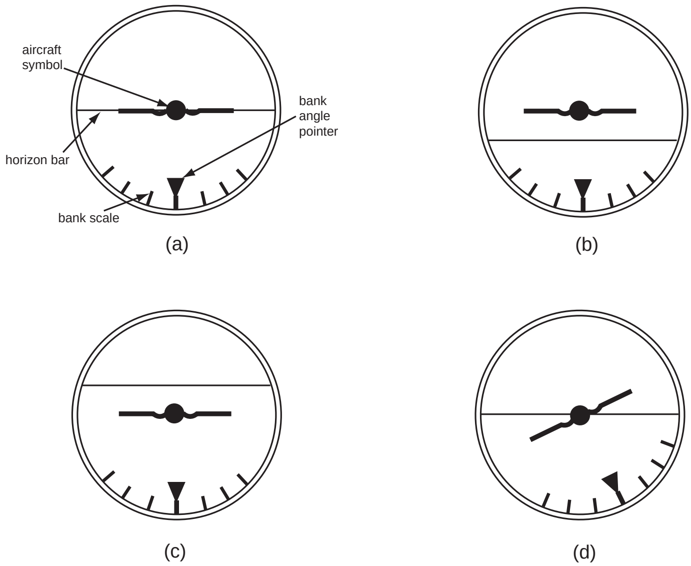

Figure 2.14 Attitude indicator presentations.

the aircraft symbol, on the glass face of the instrument, will have rolled with the aircraft about the vertically referenced gyro spin axis, which has maintained the outer gimbal, and therefore the horizon bar, earth horizontal. Typically, a pointer attached to the outer gimbal will indicate bank angle against a scale printed on the glass face of the instrument. Figure 2.14(d) shows the indication with the aircraft banked to the left.

Some attitude indicators incorporate an adjustment knob that can be used to raise or lower the aircraft symbol, so that it may be positioned against the horizon bar when the aircraft is flying straight and level, but with the fuselage pitched up or down. This is particularly useful in helicopters, which frequently fly level in a pitched attitude.

Erection mechanism

The vertical gyro of the pneumatic attitude indicator is maintained earth vertical by means of an erection unit beneath the inner gimbal. The air that has driven the gyro rotor is exhausted through four equally spaced ports machined in the sides of the unit, two in the lateral (athwartships) axis and two in the longitudinal (fore and aft) axis. When the gyro spin axis is earth vertical, each port is partly covered by a freely pivoted pendulous vane and the exhaust air escapes equally from each port, as illustrated in Figure 2.15(a).

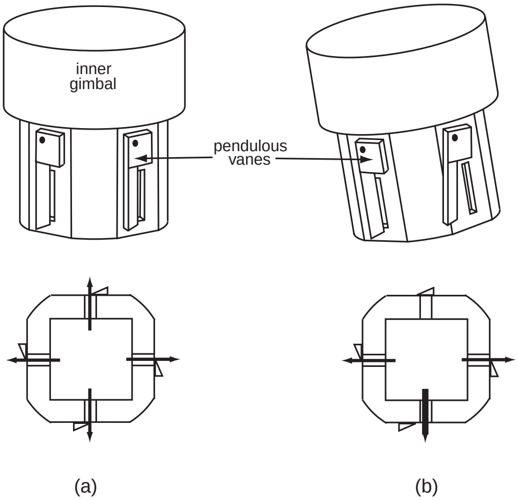

Figure 2.15 Erection system – pneumatic attitude indicator.

If the spin axis, and with it the inner gimbal, tilts away from earth vertical, the vanes, because they are pendulous, will continue to hang vertically. Suppose the gyro has tilted as shown in Figure 2.15(b). On one side of the erection unit the vane has fully uncovered its port, whilst on the other the vane will have fully covered its port. The front and rear vanes will not have moved relative to their ports, and so these ports will remain half open. Consequently, air will be exhausted from one side of the erection unit only and there will be a reaction force, in the opposite direction, applied to the gyro rotor. The gyro will precess this force 90° in the direction of rotation, which will serve to re-erect the gyroscope. When it is once again earth vertical, all four ports will be equally uncovered and the erection forces will once again be in balance.

Acceleration errors

The erection mechanism of the pneumatic attitude indicator is the cause of false attitude indications during aircraft acceleration. When the aircraft accelerates in a level attitude, such as during the take-off run, the pendulous vanes tend to swing rearward due to inertia. This does not affect the front and rear vanes. However, by referring to Figure 2.15 it will be seen that this will result in the right side port becoming uncovered more than the left side port. Because the gyro rotor spins anti-clockwise when viewed from above, the reaction to this sideways imbalance of force will apply a force to the rotor which, when precessed, will tilt the rotor to give a false climb indication.

The erection unit itself is also pendulous, suspended as it is beneath the

gyro rotor. Consequently, during a rapid acceleration inertia tends to swing it rearward, thereby applying a rearward force to the rotor. This is precessed 90° in the anti-clockwise direction of rotation to tilt the rotor to the right, giving a false indication of right bank.

Thus, the overall effect of aircraft acceleration is to give a false indication of a climbing right turn.

Because of these errors, pneumatic attitude indicators are usually only fitted to light and general aviation aircraft of low performance which have limited electrical power available. Wherever the electrical power supplies are adequate it is usual to fit electrically driven attitude indicators, which are less susceptible to acceleration error.

Turning errors

During a turn, centrifugal force acts to swing the fore and aft pendulous vanes outward from the centre of the turn. The resultant reaction force on the erection unit, when precessed, tends to tilt the gyro to give a false indication of bank. Furthermore, the pendulosity of the erection unit, suspended beneath the inner gimbal, applies a sideways force to the gyro that, when precessed, gives a false indication of pitch. In general the turning errors are less serious than the acceleration error.

Electrically driven attitude indicator

The principle of operation of the electrically driven attitude indicator is essentially the same as that of the pneumatic instrument. The gyro unit is an electric motor that rotates at considerably higher speed, typically around 22 000 rpm, and therefore has greater rigidity.

The erection system is quite different, however. Instead of the pendulous system of the pneumatic type, it employs two torque motors mounted on the outer gimbal and operated by mercury-filled levelling switches. Figure 2.16 is a diagram showing the operating principle of the system.

The torque motors are a.c. induction machines with their stators mounted on the outer gimbal in line with its lateral and longitudinal axes. When current is supplied to the stator a rotating magnetic field is set up, which tends to rotate the rotor surrounding the stator. This tendency is opposed by the rigidity of the gyroscope, resulting in a torque reaction acting about the axis of the motor, and therefore about the pitch or roll axis of the outer gimbal.

The mercury-filled levelling switches are small tubes partly filled with mercury, mounted on the inner gimbal of the gyroscope. When the gyroscope is running at normal speed an electrical current is supplied to a central contact in the tube. Contacts at either end of the tube are connected to the

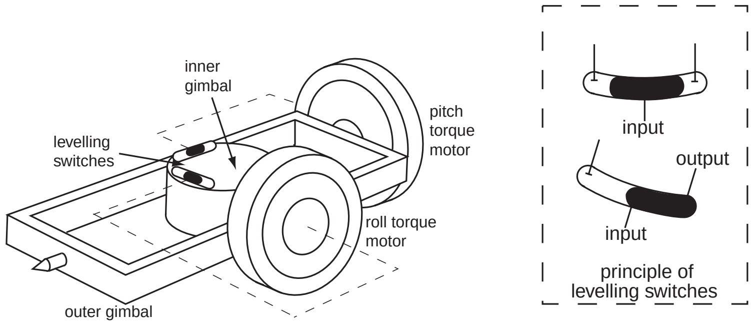

Figure 2.16 Erection system – electrical attitude indicator.

associated torque motor stator field windings. Whilst the inner gimbal remains earth horizontal the mercury in the switch is centralised and there is no conducting path between the central, supply contact and either of the end, output contacts. If the switch is tilted, due to the outer gimbal tilting, the mercury runs to one end of the tube and connects the electrical supply to one output. The direction of tilt will determine the direction of the torque applied by the appropriate torque motor. The principle is illustrated in the scrap view in Figure 2.16.

Let us suppose that the inner gimbal has begun to topple rearward, that is anti-clockwise about the pitch axis as viewed in Figure 2.16. The roll levelling switch, aligned with the pitch axis of the instrument, will not be affected and its mercury will remain centralised. The pitch levelling switch, aligned with the instrument roll axis, will be tilted and its mercury will run to the rear end of the tube, completing the electrical supply circuit to the pitch torque motor and causing it to apply a torque force anti-clockwise about the roll axis, as viewed in Figure 2.16. The torque produced will be applied to the vertical gyro and precessed through 90° in the direction of rotation. This will result in a clockwise torque force about the longitudinal axis, as viewed in Figure 2.16, acting upon the gyro to re-erect it. Once it is restored to earth vertical the levelling switches will both be in the neutral (mercury centralised) positioned and supply to the torque motors isolated.

Acceleration and turning errors

The mercury switches are susceptible to acceleration, during which inertia will force the mercury to one end of the tube and make the contacts to supply power to one or both torque motors. This would, of course, lead to false indications similar to those described for the pneumatic instrument. However, it is a relatively simple matter to incorporate a cut-out system in the electrical circuitry, which will detect acceleration and distinguish it from

topple. This will cut off supply to the switches during detected acceleration and prevent false climb or bank indication.

Warning indications

In the event of failure of the vacuum system the pneumatic attitude indicator will normally display a warning flag on the face of the instrument. The electrical attitude indicator typically displays an OFF flag when its power supply is disconnected.

Erection speed

Pneumatic attitude indicators typically have an erection speed of about 8° per minute, which means that they usually take in excess of 5 minutes to erect from start-up. Electrical attitude indicators have erection rates, of the order of 3° to 5° per minute. Many have fast erection systems for use during start-up, giving an erection time of less than one minute. Some pneumatic instruments are equipped with a caging system, similar to that described for the directional gyro, which shortens the start-up erection process.

Turn and bank indicator (rate gyro)

The purpose of the turn and bank indicator is to measure and display the aircraft rate of turn and to indicate whether the aircraft is correctly banked for a co-ordinated turn with no slip or skid. To measure the rate of turn, i.e. rate of movement about the yaw axis, the instrument employs a rate gyro that is sensitive to movement about the aircraft yaw axis only. The bank indication is a separate device using a combination of gravitational and centrifugal force.

Rate gyroscope

Since the rate gyroscope is required to be sensitive to movement about the yaw axis it follows that its spin axis must be perpendicular to that axis, i.e. horizontal. The gyro rotor is mounted in a gimbal with its spin axis aligned with the lateral (athwartships) axis of the aircraft. The single gimbal is pivoted fore and aft in the instrument casing, in line with the aircraft longitudinal axis. The gyro rotor spins up and away from the pilot. The general arrangement showing the principle of operation is shown in Figure 2.17. It will be seen that the gyro has freedom of movement about two axes only, the lateral spin axis and the longitudinal precession axis.

When the aircraft yaws about the vertical axis this applies a force to the gyro rotor at the front, in line with the spin axis. Let us suppose that the

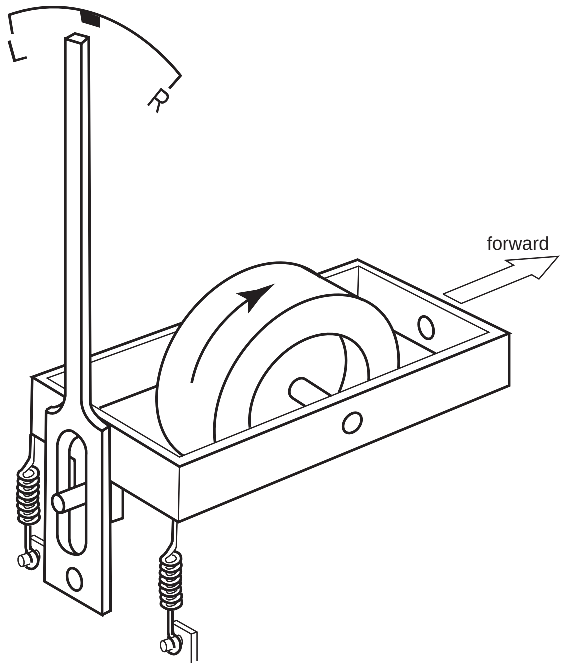

Figure 2.17 Turn and bank indicator – principle of operation.

aircraft is turning to the left. This applies a torque force about the yaw axis in an anti-clockwise direction viewed from above. This is as though a linear force were applied to the front of the gyro rotor on the right side in line with the spin axis, as illustrated at Figure 2.17.

The gyro will precess this force 90° in the direction of rotation, so that it becomes torque acting in a clockwise direction about the longitudinal axis, precessing the gyro so that the gimbal begins to tilt to the right. The extent to which the gimbal tilts is limited by a spring connecting the gimbal to the instrument casing. As the spring is stretched it exerts a force on the gimbal opposing the precession. When the two are in balance the gimbal is held at a tilt angle that is proportional to the rate of turn, because the precession is equal to the rate of turn and the angular momentum of the gyroscope.

Thus, the greater the rate of turn, the greater the tilt of the gimbal. The gimbal actuates a pointer, which moves against a calibrated scale on the face of the instrument to indicate rate of turn. The actuation is such that when the gimbal tilts to the right the pointer moves to the left and vice versa.

The speed of rotation of the turn indicator gyro is relatively low, typically about 4500 rpm. It is critical that its speed is maintained constant, since this is a vital factor in ensuring that precession remains constant relative to rate of

turn. A warning flag will appear on the face of the instrument when the gyro rotational speed is outside limits.

Rate of turn is classified numerically, where rate 1 equals 180° per minute, rate 2 equals 360° per minute, rate 3 equals 540° per minute and rate 4 equals 720° per minute. These may also be quoted as 3°, 6°, 9° and 12° per second, respectively. An aircraft maintaining a rate 1 turn for 2 minutes will therefore turn through 360°.

Bank indication

The bank indication given by the turn and bank indicator displays to the pilot whether or not the aircraft is correctly banked for the turn being made. If the aircraft is banked excessively it will tend to slip toward the centre of the turn, whereas if it is underbanked it will skid outwards, away from the centre of the turn. Hence the name by which this instrument was once commonly known, the turn and slip indicator.

The display is provided by a device quite separate from the rate gyroscope of the turn indicator, and typically comprises a curved glass tube filled with liquid and containing a ball. When the aircraft is in level flight, gravity ensures that the ball lies in the centre of the curved tube, as shown in Figure 2.18(a). When the pilot is making a properly co-ordinated banked turn the glass tube, which is attached to the instrument, will be banked with the aircraft and the resultant of centrifugal force and gravitational force will keep the ball in the centre, as shown at Figure 2.18(b). Suppose now that the aircraft is turning, but that the bank angle is greater than it should be, i.e. the aircraft is overbanked. The centrifugal force acting on the ball is less than the gravitational force and the ball falls into the lower part of the tube, as shown in Figure 2.18(c). This indicates to the pilot that the aircraft is slipping into the turn. If the aircraft is underbanked the centrifugal force acting on the ball is greater than the gravitational force and the ball will be moved into the upper part of the tube, indicating that the aircraft is skidding out of the turn. This is shown in Figure 2.18(d). Figure 2.18(e) shows the turn and bank indications during a properly co-ordinated 2 minute (rate 1) standard turn.

Turn co-ordinator

Light aircraft are often fitted with a variation of the turn and bank indicator, known as a turn co-ordinator. The purpose of the instrument is to present the pilot with a display that makes co-ordination of bank angle and turn rate as simple as possible. The display is as shown in Figure 2.19. When the pilot banks the aircraft to initiate a turn the aircraft symbol on the display banks in the appropriate direction, since it is actuated by the gyro precession exactly as previously described. Provided that the aircraft

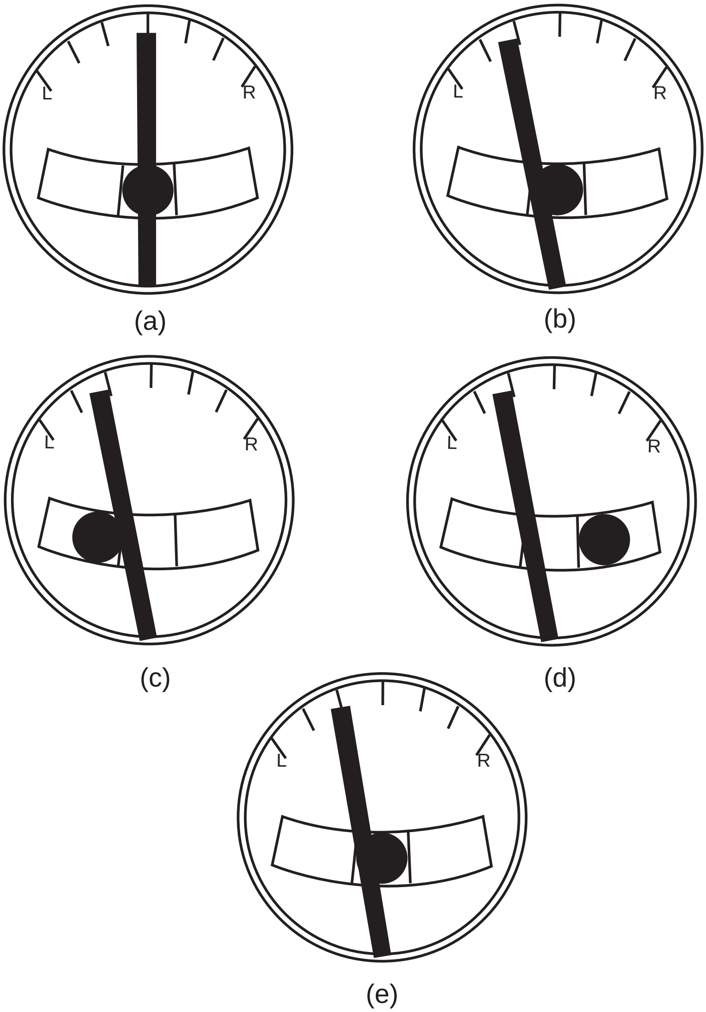

Figure 2.18 Turn and bank indications.

symbol is aligned with the left or right bank indication on the display, and the ball is in the centre, the aircraft will be making a properly banked rate 1 (2-minute) turn.

The main constructional difference between this and the turn and bank indicator is that the longitudinal axis of the gyro gimbal is inclined at 30° to the horizontal, so that the gyro will respond to banking as well as turning input force.

The movement of the gimbal ring of all indicators is damped to control the rate of precession. Among other effects, this will limit the instrument bank indication when turning during ground taxiing.

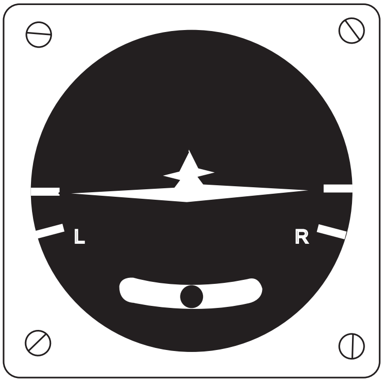

Figure 2.19 Turn co-ordinator display.

Magnetic compass

The magnetic compass works on the principle that a freely suspended magnet will align itself with the earth's magnetic field such that one end will point toward the north magnetic pole. It is mandatory that all civil aircraft must carry such a compass and in all but light general aviation aircraft it serves as a standby compass.

To be effective a magnetic compass must meet three basic requirements.

Horizontality

The compass card must lie horizontal, since it is direction parallel to the earth's surface that it is required to indicate. This is not quite as simple as it might sound, since the earth's magnetic field dips toward the poles and is only parallel to the surface near the equator. This is illustrated in Figure 2.20.

For simplicity, the magnetic force is resolved into its horizontal and vertical components, H and Z respectively. Thus, the nearer one is to either terrestrial magnetic pole, the greater the Z force and the weaker the H force. This causes the compass magnet to dip toward the earth's surface and, clearly, the greater the angle of dip, the less accurate the compass becomes.

The amount by which the compass magnet system dips can be reduced significantly by suspending it so that the centre of gravity of the magnet system is well below the pivot point of the circular plate to which the magnets and the compass card are attached. The principle is illustrated in Figure 2.21, from which it can be seen that the tilt caused by the Z force effect is countered by the CG becoming offset from the vertical pivot line and the angle of dip is the resultant of the Z force and gravity.

Figure 2.21 shows the effect of a lowered CG on dip angle in the northern hemisphere, where the magnet system dips toward the north magnetic pole. In the UK the dip angle is about 2° to 3°, but the further north the compass is

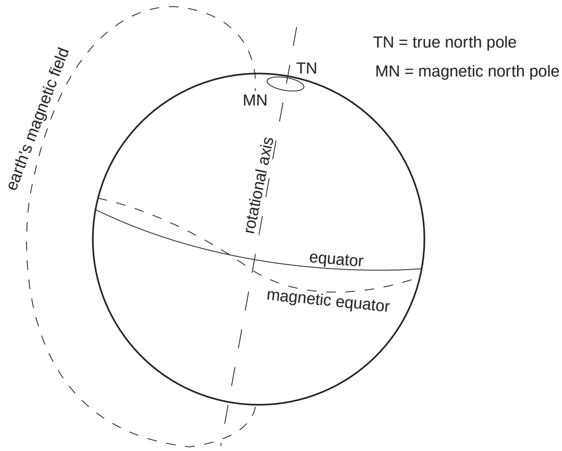

Figure 2.20 Terrestrial magnetism.

taken the greater this will become until, at about 70°N the compass becomes unusable. In the southern hemisphere the magnet system dips toward the south magnetic pole and the compass again becomes useless above about 70°S. Hence, in the northern hemisphere the magnet system CG lies to the south of the pivot point and in the southern hemisphere it lies to the north of the pivot point. This is of significance when we come to consider the effects of turning and acceleration on the compass.

N.B.: Figure 2.21 is descriptive only. It should be noted that in reality the suspended magnets would not appear as shown, but would be aligned with the E–W cardinal points of the compass card.

Sensitivity

It is essential that the magnet system of the compass shall point firmly along the magnetic meridian toward the north magnetic pole, and this is achieved by using magnets of sufficient pole strength. To ensure that they continue so to do when the aircraft heading changes, friction at the pivot point is minimised by using a jewelled bearing with an iridium pivot. Furthermore,

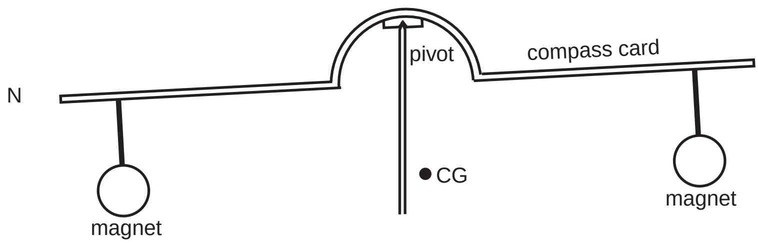

Figure 2.21 Magnet suspension system.

the compass system is suspended in a clear liquid, which lubricates the bearing.

Aperiodicity

When a compass is deflected away from its north seeking direction, it is desirable that it will return to that direction as quickly as possible. If it tends to oscillate about north for a significant period of time before once again coming to rest it is said to be *periodic*. Ideally, it should come to rest with no oscillations, when it would be said to be *aperiodic*. Aperiodicity is achieved in aircraft magnetic compasses by three measures:

- The interior of the compass case, known as the bowl, is filled with a clear liquid, typically silicone or methyl alcohol. The compass card has filaments attached to it and these provide a damping effect in the liquid to minimise oscillation.
- Instead of a single powerful magnet, several small powerful magnets are positioned close to the pivot point and this reduces the moment of inertia of the magnet system.
- The suspension of the magnet system in fluid helps support the apparent weight of the magnet system, further reducing the moment of inertia.

Provision is made for the expansion and contraction of the compass liquid by fitting a bellows, or in some cases a flexible diaphragm, inside the compass bowl. The construction of a typical aircraft standby compass is shown in Figure 2.22.

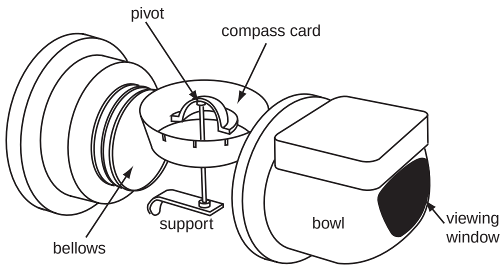

Figure 2.22 Typical standby compass.

Turning and acceleration errors

We have already seen how the magnet system of the compass dips at northern or southern latitudes, displacing the CG of the system from the pivot point. This displacement has a profound effect upon the compass magnet system during aircraft turns, accelerations and decelerations. Let us

first consider the effect of acceleration and deceleration on a compass in the northern hemisphere, using Figure 2.23 as a reference.

Acceleration error

Figure 2.23(a) shows the compass viewed from above, with the CG displaced southward of the pivot point and the compass card markings simplified to show only the cardinal points. In Figure 2.23(b) the aircraft is accelerating on a westerly heading. Inertia causes the suspended magnet system to lag and, because its CG is displaced from the pivot point the moment of inertia causes the compass card to rotate anti-clockwise. To the pilot this presents a false indication of a turn toward north. If the aircraft were to decelerate, the lag would be in the opposite direction and the compass would swing to give a false indication of a turn toward south.

In Figure 2.23(c) the aircraft is accelerating on an easterly heading and the lag of the displaced CG causes the compass card to swing in a clockwise direction. Once again, the effect as far as the pilot is concerned is that the compass is falsely indicating a turn toward north. Deceleration on the same heading would produce a false indication of a turn toward south.

Acceleration or deceleration on a northerly or southerly heading will not cause false indications, since the CG and pivot point are aligned.

Summarising, in the northern hemisphere acceleration on an easterly or westerly heading will produce a false indication of a turn toward north; deceleration on those headings will produce a false indication of a turn toward south.

In the southern hemisphere the errors are the opposite of the above.

Turning errors

Turning errors due to the displacement of the CG are maximum on a northerly or southerly heading and are significant up to 35° on either side of those headings. Let us consider the situation with an aircraft in the northern

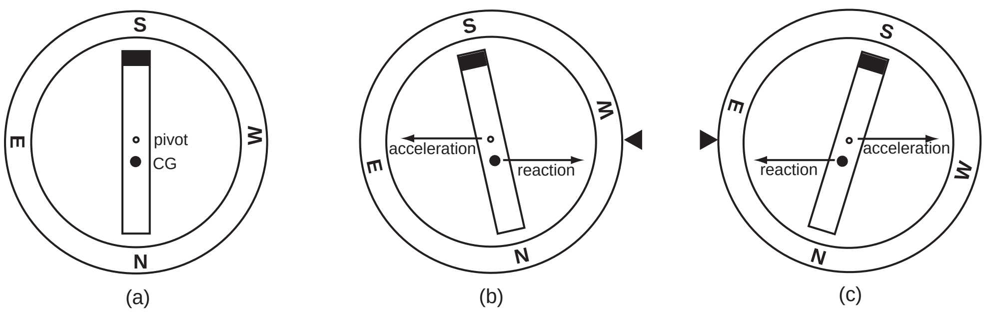

Figure 2.23 Acceleration error.

hemisphere and turning through a northerly heading. Figure 2.24(a) illustrates the situation when a turn toward east is initiated.

As the aircraft turns, the centrifugal force acting upon the CG of the magnet system creates a turning force on the compass card in the same direction as the turn. The card should not turn at all, of course, but should remain aligned with the magnetic meridian. The effect of the turning card is to cause the compass to underread during the turn. A turn through north toward west will have the same effect, the card turning in the same direction as the aircraft turn and causing the compass to underread the turn. This is illustrated in Figure 2.24(b).

Figure 2.24(c) shows the situation with the aircraft turning toward east through a southerly heading. Now the centrifugal force acting on the displaced CG turns the compass card in the opposite direction to the turn and the compass will overread during the turn. Exactly the same will happen when a turn toward west is made through a southerly heading, as illustrated in Figure 2.24(d).

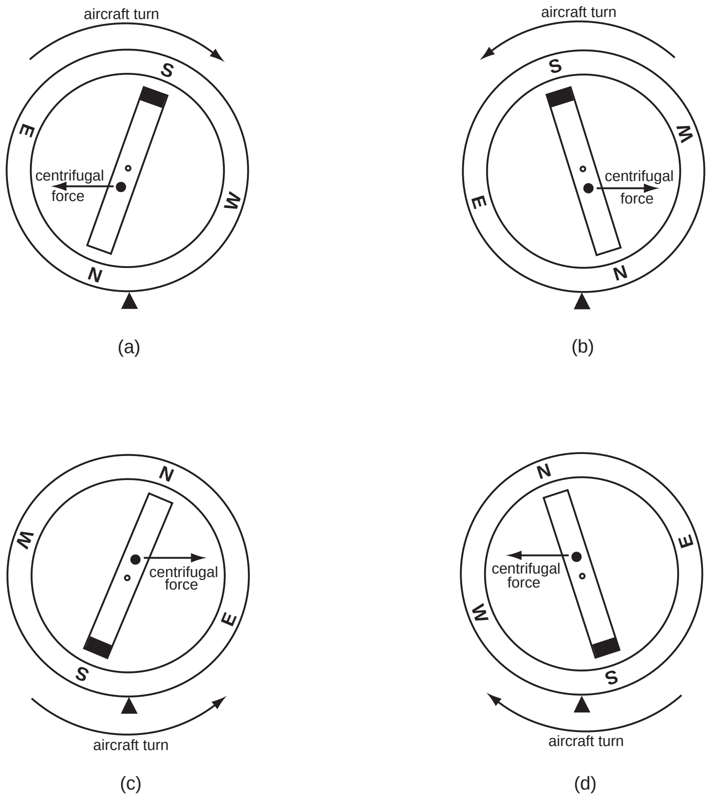

Figure 2.24 Turning errors.

In the southern hemisphere the turning errors are opposite to those described above. The overall effects are summarised in Table 2.1.

Table 2.1 Turning errors.

Turn through	Turn toward	Northern hemisphere	Southern hemisphere
		Compass error	**Compass error**
NORTH	EAST	UNDERREAD	OVERREAD
NORTH	WEST	UNDERREAD	OVERREAD
SOUTH	EAST	OVERREAD	UNDERREAD
SOUTH	WEST	OVERREAD	UNDERREAD

When turning through east or west there is no turning error, since the CG and pivot point are in alignment. When turning with the magnetic compass as reference it is necessary to roll out of the turn early (i.e. before the new desired heading is reached) when it is underreading and late when it is overreading.

Variation and deviation

Variation

Referring back to Figure 2.20, it will be seen that the earth's true and magnetic north poles are not co-incident and there is, in fact, some considerable geographic distance between the two. At most locations there will be an angular difference between the magnetic north and true north, and this difference is known as magnetic variation. On the shortest distance line joining the true and magnetic poles, variation at any point is theoretically 180°, whereas elsewhere on a line joining the two, known as an agonic line, variation is zero.

At locations where magnetic north lies to the east of true north, variation is said to be easterly. At points where magnetic north lies to the west of true north, variation is said to be westerly. The concept is illustrated in Figure 2.25.

Magnetic variation at any point on the earth's surface can be plotted and is shown on charts as a series of lines joining points of equal variation, known as isogonals. The earth's magnetic field undergoes various changes which, in the long term, cause the location of the magnetic poles to move. This movement is reasonably predictable and usually appears on charts as annual changes to variation, e.g. annual change 5′W.

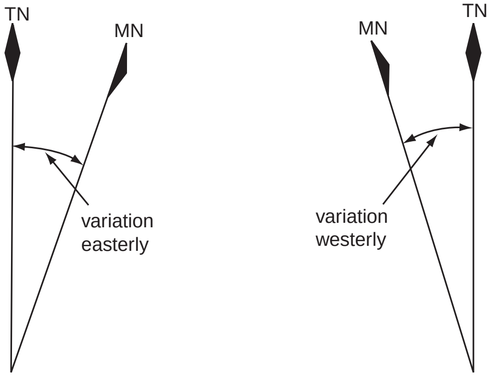

Figure 2.25 Magnetic variation.

Deviation

A compass needle will indicate the direction of magnetic north, provided that it is only influenced by the earth's magnetic field. In an aircraft there are many influencing local magnetic fields caused by hard or soft iron or electrical circuits. These will deflect the compass needle away from the direction of magnetic north. The angular difference between compass north and magnetic north is referred to as deviation.

The strength of the aircraft's magnetic field is usually reasonably constant and its effect on the compass reading can be determined during a procedure known as compass swinging. The aircraft is placed on a number of known magnetic headings and the compass reading is compared with the aircraft heading, to find the angular difference between compass north and magnetic north. The compass reading can be adjusted by means of a compensation device within the instrument until it agrees as nearly as possible with the aircraft heading. It is usually impossible to adjust the compass so that it exactly agrees on all headings and the remaining small angular difference is known as residual compass deviation.

Upon completion of the compass swing the residual compass deviation is recorded on a compass correction card, in either tabular or graphic form, which is mounted in the cockpit. The correction card shows the deviation as positive (+) or negative (−), indicating how it must be applied to the compass reading to obtain the correct magnetic heading.

When compass north lies to the east of magnetic north, deviation is said to be easterly; when it lies to the west of magnetic north, deviation is said to be westerly. Deviation easterly is positive and deviation westerly is negative. Thus, if the correction card states that the compass has -1° residual deviation on a heading of 180°(M), then the pilot must steer 181°(C) for the aircraft to actually be on a heading of 180°(M). Alternatively, when the

compass reads 180°(C) the aircraft's magnetic heading will be 179°(M). The concept is illustrated in Figure 2.26.

The maximum permissible value for residual deviation of a standby compass is $\pm 10°$.

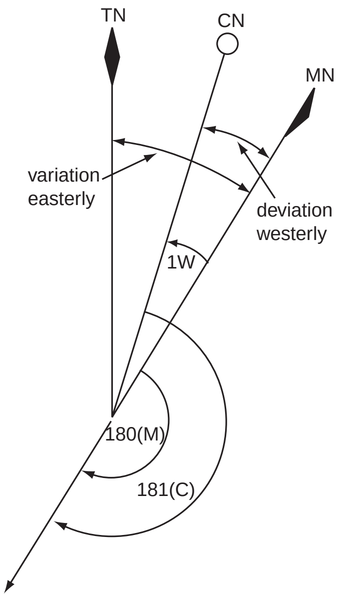

Figure 2.26 Compass deviation.

Slaved gyro compass

- The slaved gyro compass, also known as a remote indicating compass or gyro-magnetic compass, embodies the best features of the directional gyro and the direct reading compass.
- The direct reading compass suffers from a number of disadvantages, in particular its susceptibility to acceleration and turning errors. Because it must be located at the position where it is to be read, e.g. the cockpit, it is susceptible to aircraft magnetism and suffers from deviation.
- The directional gyro is subject to apparent drift when it is at any location other than that for which it is adjusted and consequently requires constant re-setting.
- The slaved gyro compass utilises the earth's magnetic field to sense magnetic north and this magnetic flux is used to provide the constant correction required by the gyroscopic element of the compass system to maintain a magnetic north reference. The gyroscopic stabilisation reduces the turning and acceleration errors inherent in the magnetic sensing ele-

ment and the device has the further advantage of being capable of operating any number of remote compass indicators.

The fluxvalve

The magnetic detecting element of the slaved gyro compass is a device known as a fluxvalve. This senses the direction of the earth's magnetic field relative to the aircraft heading and converts this into an electrical signal. A simple fluxvalve comprises two bars of highly permeable magnetic material that are laid parallel to each other, each surrounded by a coil of wire, called a primary coil, supplied with alternating current and connected in series. Surrounding these is a secondary, pick-up coil. The general arrangement is illustrated in Figure 2.27.

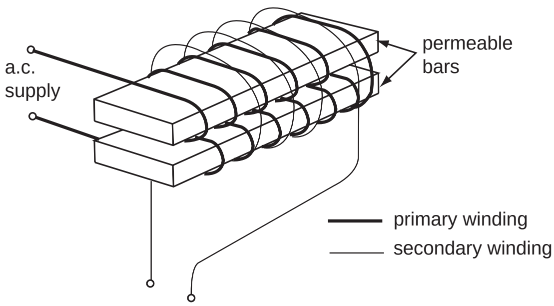

Figure 2.27 Simple fluxvalve.

When a current is passed through a coil of wire a magnetic field is set up around the coil and, if the current is alternating current, the field will be of continuously varying strength and polarity. Because the primary coils are wound in opposite directions, the fields surrounding them are of opposite polarity and the permeable fluxvalve bars are magnetised with opposite polarity by the primary fields. The alternating current is of sufficient strength that, at its peak value, both bars are magnetically saturated.

A variable magnetic flux field cutting through the windings of the secondary coil would normally induce a current flow in them, but because the primary fields are of opposite polarity and therefore self-cancelling, no current is induced in the secondary coil.

However, if the fluxvalve is placed parallel to the earth's surface it will be subjected not only to the electro-magnetic field due to the primary current flow, but also to the horizontal (H) component of the earth's magnetic field. This flux due to the earth's magnetic field will saturate the bars of the fluxvalve before the alternating current supply to the primary coils reaches

peak value and will cause a varying strength of flux around the primary coils.

As this varying flux cuts through the windings of the secondary coil, it will induce a current flow in the secondary windings. The strength of this current is directly proportional to the strength of flux in the fluxvalve bars due to earth magnetism. This will depend upon the direction of the bars relative to the earth's magnetic field, which will be greatest when the bars lie parallel to the earth's (H) field and weakest when they are at right angles to it. The output of the secondary coil can be used to drive a remote indicating compass or, in the case of the slaved gyro compass, to apply corrections to the gyro unit.

Detector unit

The detector unit contains the fluxvalve that senses the direction of the earth's magnetic field relative to the aircraft heading. However, a simple fluxvalve of the type described above would be inadequate, since it is prone to ambiguity in that it will produce a signal of identical strength and polarity on different headings. In order to overcome this anomaly an aircraft sensing unit consists of three simple fluxvalves connected at a central point and spaced at 120° intervals, as shown in Figure 2.28. Each fluxvalve has a collector horn attached to its outer end to improve collection of the relatively weak earth flux. The assembly is pendulously suspended so that it will remain horizontal regardless of aircraft attitude and is mounted where it will be least affected by aircraft magnetism, typically at the wingtip or near the top of the fin (vertical stabiliser).

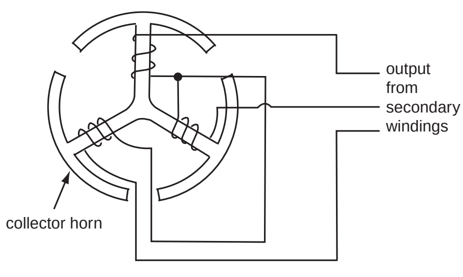

Figure 2.28 Detector unit.

Remote compass indication

The detector unit is mounted such that it remains essentially earth horizontal, in order to best sense the H component of earth magnetism, and fixed so that one 'spoke' of the detector is permanently aligned with the

aircraft's longitudinal axis. Each spoke has its primary and secondary windings, and the secondary windings are connected to the stator windings of a device called a signal selsyn, as illustrated in Figure 2.29. The signal selsyn comprises three stator windings, mounted at 120° to each other, surrounding a rotor upon which is wound a further coil.

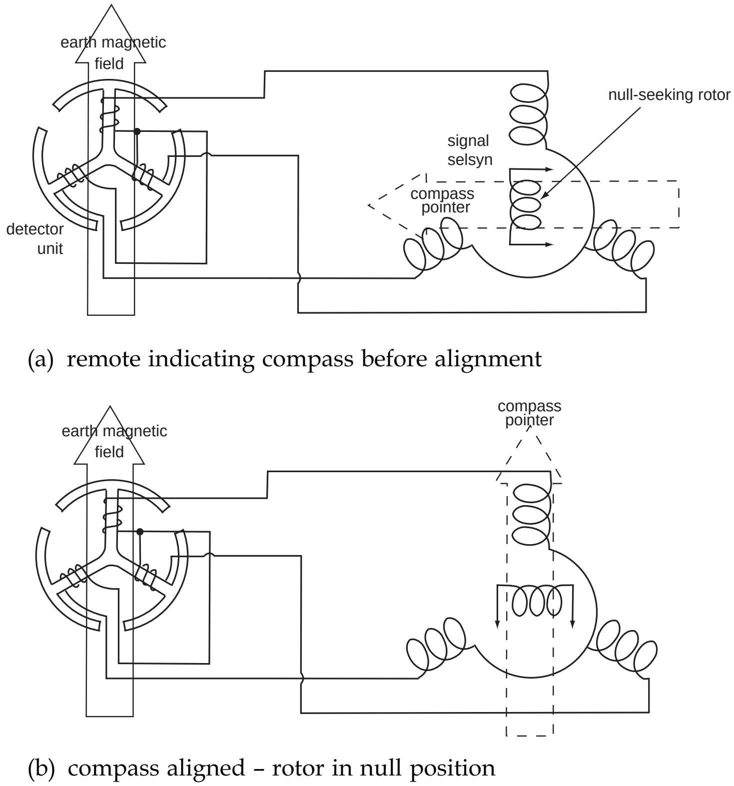

Figure 2.29 Remote compass indication.

The following is a description of the principle of operation of a remote indicating compass system. Let us suppose that the aircraft is on a heading of 000°M. Coil A of the detector unit is aligned with the earth H field and therefore maximum current is induced in it. Coils B and C have weaker strength current induced in them and, because of the direction of their windings, it is of opposite polarity. These currents are supplied to coils A, B and C of the signal selsyn and the combination of their electro-magnetic fields reproduces a flux field identical to that sensed by the detector unit.

Assume for the moment that the rotor of the signal selsyn is aligned with this reproduced field. The lines of flux cutting through the windings of the rotor coil will induce maximum voltage and current flow within the coil.

This current flow creates an electro-magnetic field, which will seek to align with the reproduced 'earth' magnetic, causing the rotor to rotate until it is at right angles to the reproduced field. When this is reached there will be no voltage induced in the rotor coil and the motor will cease turning, with the compass pointer indicating the heading as sensed by the detector unit. This is known as the null point of the signal selsyn rotor. Attached to the rotor is a compass pointer, indicating the aircraft magnetic heading.

Because the fluxvalves of the detector unit are at 120° to each other their combined secondary coil outputs are unique on every heading. This ensures that the null-seeking rotor will always be at right angles to the selsyn reproduced 'earth' magnetic field.

As aircraft heading changes, the relationship of the detector unit flux-valves to the earth's magnetic field will change with it. This field will be reproduced in the signal selsyn and the rotor will turn to maintain itself at right angles to the field, moving the compass pointer so that the heading indication changes with the heading change.

In the case of the slaved gyro compass a signal from the null-seeking rotor is used to precess a gyroscope to the correct magnetic heading reference. A schematic diagram of a simple slaved gyro compass system is in Figure 2.30.

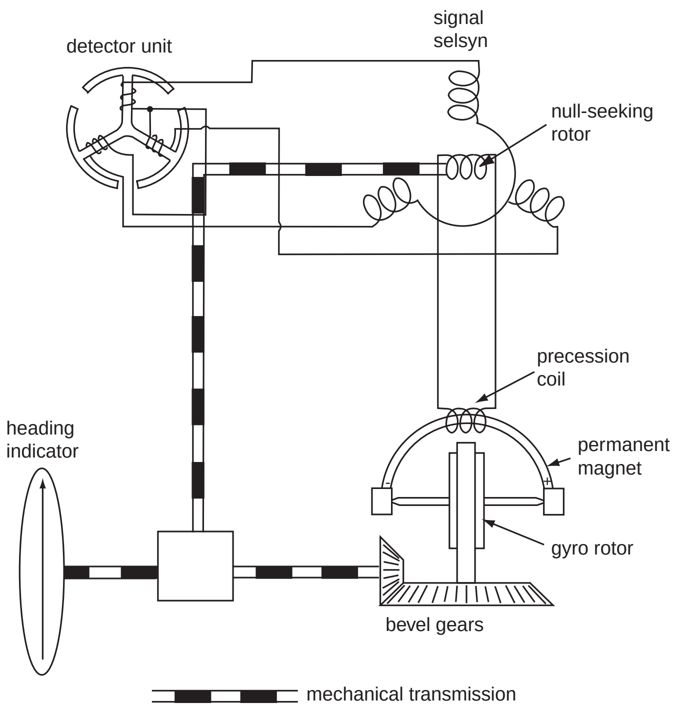

Figure 2.30 Slaved gyro compass system schematic.

Gyro compass operation

The gyroscope of the slaved gyro compass is basically similar to that of the directional gyro. Its horizontal spin axis is mounted within an inner gimbal, which is in turn pivoted to an outer gimbal that has freedom of movement about the vertical axis. Attached to the inner gimbal is a permanent magnet, as shown in Figure 2.30.

A precession coil is wound around the permanent magnet and is supplied with direct current (d.c.) from the null-seeking rotor coil. When current flows through the precession coil a magnetic field will be produced, the polarity of which will depend upon the direction of the current flow. This magnetic field will react with that of the permanent magnet, applying a force to it that tends to tilt the spin axis of the gyroscope.

However, the gyroscope will precess this force through 90° in the direction of rotation, which will cause the gyro to rotate about its vertical axis. It will continue to do this until the current flow to the precession coil ceases, which will happen when the null-seeking rotor is at right angles to the selsyn reproduced magnetic field. Thus, aircraft heading changes will generate a current flow from the null-seeking rotor, precessing the gyro to align itself to the new heading.

The gyro is mechanically connected through bevel gears to the compass display pointer and to the null-seeking rotor. This latter ensures that any tendency of the gyro to drift will move the rotor from its null position, causing a current to be generated in the rotor coil. This will be transmitted to the gyro precession coil, precessing the gyro to keep it aligned with the aircraft magnetic heading. Apparent drift due to earth rotation or transport wander is thereby eliminated.

As with the directional gyro, the inner gimbal of the gyro compass gyro must be maintained horizontal and to achieve this an erection system is required. In this case the erection system consists of a torque motor mounted on the outer gimbal, activated by a levelling switch mounted on the inner gimbal.

Gyro compass errors

Since the detector unit of the slaved gyro compass is pendulously mounted, it follows that during aircraft accelerations the unit will tilt and sense the vertical (Z) component of earth magnetism. In order to minimise the errors that would otherwise arise during turning and acceleration, the precession rate of the gyroscope is deliberately kept slow. Thus, during an aircraft acceleration the heading reference is maintained by the gyro, although the rotor coil may be transmitting a false signal to the precession coil. Errors due to detector unit tilt during a turn only become significant if a slow rate of

turn is maintained for a significant period of time. During a normal turn the error is very small due to the slow precession rate of the gyro. Deviation errors are small in the slaved gyro compass because the detector unit is mounted as far as possible from deviating magnetic influences. Compensation for those small remaining influences may be made by means of a compensator unit that produces small electro-magnetic fields to oppose those causing deviation.

Gyro compass outputs

Output from the slaved gyro compass may be used to supply magnetic heading information to the radio magnetic indicator (RMI), the horizontal situation indicator (HSI) of a flight director system, the autopilot system and navigation systems such as INS and Doppler.

Sample questions

1. The two basic properties of a gyroscope are rigidity and . . . .:

   a. Aperiodicity?
   b. Precession?
   c. Rotation?
   d. Sensitivity?

2. Concentrating the mass of a gyro rotor at the rim:

   a. Increases the rate of precession?
   b. Decreases its rigidity?
   c. Decreases its rate of precession?
   d. Reduces its angular momentum?

3. A free gyro is one that:

   a. Has two gimbals and freedom of movement about three axes?
   b. Has one gimbal and freedom of movement about two axes?
   c. Has two gimbals and freedom of movement about two axes?
   d. Has three gimbals and freedom of movement about three axes?

4. When the spin axis of a gyroscope deviates from its fixed reference due to manufacturing imperfections this is known as:

   a. Apparent drift?
   b. Apparent wander?
   c. Apparent topple?
   d. Real wander?

5. The rate of apparent drift due to earth rotation is given as:

   a. 15°/hr?
   b. 15 × sin latitude °/hr?
   c. 15 × cos latitude °/hr?
   d. Sin latitude °/hr?

6. An advantage of the ring laser gyro over a conventional gyro is that it:

   a. Is cheaper to produce?
   b. Has greater freedom of movement?
   c. Has higher spin speed?
   d. Is immediately available for use when switched on?

7. A directional gyro is most likely to suffer from gimbal error:

   a. When manoeuvring on headings other than cardinal headings?
   b. When flying straight and level on cardinal headings?
   c. When manoeuvring on cardinal headings?
   d. When in a steady climb or descent on a non-cardinal heading?

8. An aircraft is stationary on the ground at latitude 55°N. The apparent drift rate suffered by its directional gyro will be:

   a. 12.3°/hr increasing?
   b. 8.6°/hr increasing?
   c. 12.3°/hr decreasing?
   d. 8.6°/hr decreasing?

9. Compensation for apparent drift in the directional gyro is made by:

   a. Eliminating manufacturing imperfections?
   b. Adjustment of the latitude nut?
   c. Imbalance of the outer gimbal?
   d. Increasing the rotational speed?

10. The attitude indicator uses:

    a. An earth gyro spinning at about 4500 rpm?
    b. A free gyro spinning at about 12 000 rpm?
    c. A tied gyro spinning at about 25 000 rpm?
    d. An earth gyro spinning at about 15 000 rpm?

11. During acceleration a pneumatic attitude indicator will give a false indication of:

    a. Descending left turn?

b. Climbing left turn?
c. Climbing right turn?
a. Descending right turn?

12. The erection system of an electrical attitude indicator uses:

a. Mercury switches on the inner gimbal and torque motors on the outer gimbal?
b. Mercury switches on the outer gimbal and torque motors on the inner gimbal?
c. Mercury switches and torque motors on the inner gimbal?
d. Mercury switches and torque motors on the outer gimbal?

13. The turn and bank indicator uses:

a. An earth gyro?
b. A free gyro?
c. A vertical gyro?
d. A rate gyro?

14. An aircraft is making a rate 1 turn through 180°. The turn will take:

a. 2 minutes?
b. 1 minute?
c. $1\frac{1}{2}$ minutes?
d. 3 minutes?

15. The principal constructional difference between the turn co-ordinator and the turn and bank indicator is that:

a. The turn co-ordinator does not require bank indication?
b. The turn co-ordinator has two gimbals whilst the turn and bank indicator has only one?
c. The longitudinal axis of the gyro gimbal is inclined at 30° to the horizontal?
d. The turn co-ordinator display does not use an aircraft symbol?

16. The three basic requirements of a direct-reading magnetic compass are:

a. Horizontality, rigidity and sensitivity?
b. Sensitivity, rigidity and aperiodicity?
c. Horizontality, sensitivity and aperiodicity?
d. Aperiodicity, sensitivity and rigidity?

17. An aircraft in the northern hemisphere is accelerating on an easterly heading. Its standby magnetic compass will:

a. Indicate a turn toward north?
b. Indicate a turn toward south?
c. Overread?
d. Be unaffected on this heading?

18. An aircraft flying in the southern hemisphere is making a turn toward east through a southerly heading. Its standby magnetic compass will:

a. Overread during the turn?
b. Underread during the turn?
c. Indicate a turn toward north?
a. Be unaffected on this heading?

19. An aircraft compass deviation card shows that the residual deviation on a heading of 135° is 2° easterly. In order to fly a heading of 135°M, the pilot must steer a compass heading of:

a. 135°?
b. 133°?
c. 130°?
d. 137°?

20. The maximum permissible residual deviation for a direct reading standby magnetic compass is:

a. ±5°?
b. ±3°?
c. ±10°?
d. ±8°?

21. The output signal of a fluxvalve is generated in:

a. The primary windings?
b. The permeable metal bars?
c. The secondary winding?
d. The collector horns?

22. The detector unit of a slaved gyro compass:

a. Is pendulously mounted, but fixed in azimuth?
b. Is pendulously mounted and free to move in azimuth?
c. Is free to move in azimuth because it is north-seeking?
d. Is designed to sense the Z component of earth magnetism?

23. The output of the null-seeking rotor of a slaved gyro compass will be maximum when it is:

a. At right angles to the signal selsyn field?
b. Aligned with the earth magnetic field?
c. Aligned with the signal selsyn magnetic field?
d. Aligned with the gyro heading?

Chapter 3
Inertial Navigation Systems

Inertial navigation systems are computer-based self-contained systems that provide aircraft geographic position information in terms of latitude and longitude, together with aircraft speed, heading and tracking information. When provided with a TAS input, the system also produces an output of wind velocity and direction. They require no external information or reference other than the starting location of the aircraft.

The basis of the inertial navigation system lies in measurement of the aircraft's acceleration in a known direction and this is accomplished with the use of accelerometers. These are devices that measure acceleration along a specific axis; normally one measures accelerations and decelerations along the east–west axis and a second measures accelerations and decelerations along the north–south axis. Acceleration may be defined as increase of velocity per unit time and is usually expressed in terms of metres or feet per second per second (m/s^2 or ft/s^2).

If a vehicle, such as an aircraft, accelerates from rest or steady speed at a constant rate over a given period of time, its final velocity and the distance travelled can be calculated from simple formulae:

$$v = u + at$$

and

$$s = ut + \tfrac{1}{2}at^2$$

where: v = final velocity
u = initial velocity
a = acceleration
t = time
s = distance travelled

Since aircraft accelerations and decelerations are seldom constant it becomes necessary to integrate each acceleration with respect to time in order to obtain velocity and then to integrate the result of that with time in order to obtain distance travelled. To achieve this, the outputs from the accelerometers are fed to two integrators in series, as shown in Figure 3.1.

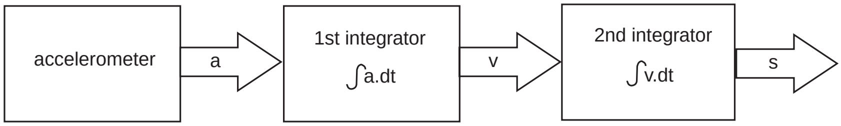

Figure 3.1 Calculation of speed and distance travelled.

In order to determine position from the factors known, speed, distance travelled and start position, it is necessary to know the direction of travel. This is determined by virtue of the two accelerometers being aligned east–west and north–south. Suppose, for example, that the north–south accelerometer/integrator combination has recorded a distance travelled of 60 nautical miles (nm) and the east–west accelerometer/integrator combination has recorded zero distance travelled. Clearly the aircraft is now 60 nm, or 1° of latitude, north of its previous position. If both the east–west and north–south accelerometers have recorded speed and distance, then the aircraft is at some point at a known distance and in a calculable direction from its start point.

In order for the system to work, the accelerometers must only measure aircraft accelerations and, to do this, they must be maintained earth horizontal at all times so that they do not measure the acceleration due to gravity (9.81 m/s^2 or 32.2 ft/s^2).

Accelerometers can be maintained physically horizontal to the earth on a gyro-stabilised platform called an Inertial Navigation System (INS). Alternatively, the accelerometers can be fixed to the aircraft axes, in which case the accelerations due to gravity and aircraft manoeuvres are removed mathematically from the accelerometer outputs. This system, called a strapdown inertial system, is the basis of an Inertial Reference System (IRS).

We shall first study the gyro stabilised platform inertial navigation system.

Accelerometers

Since the accelerometers are the heart of the inertial navigation system it should come as no surprise that they use inertia to measure acceleration. There are a number of different types of accelerometer, but the one most commonly used in aircraft inertial systems is the pendulous force balance type.

If a freely suspended pendulum is subjected to acceleration it will lag, due to inertia, in a direction opposite to that of the acceleration. The accelerometer incorporates a pendulous mass that is constrained so that it responds only to acceleration or deceleration along its sensitive axis.

In the example illustrated in Figure 3.2 the pendulous mass is suspended by two light leaf springs between two pick-off transformers. Ferrite armatures are attached to the ends of the mass and the transformer primary coils are supplied with low voltage alternating current at a frequency of about 12 kHz. Whilst the accelerometer is not subjected to acceleration, the suspended mass is positioned exactly midway between the two transformers. Under this condition the secondary voltage is identical in both transformers and there is consequently no current flow in the secondary circuit.

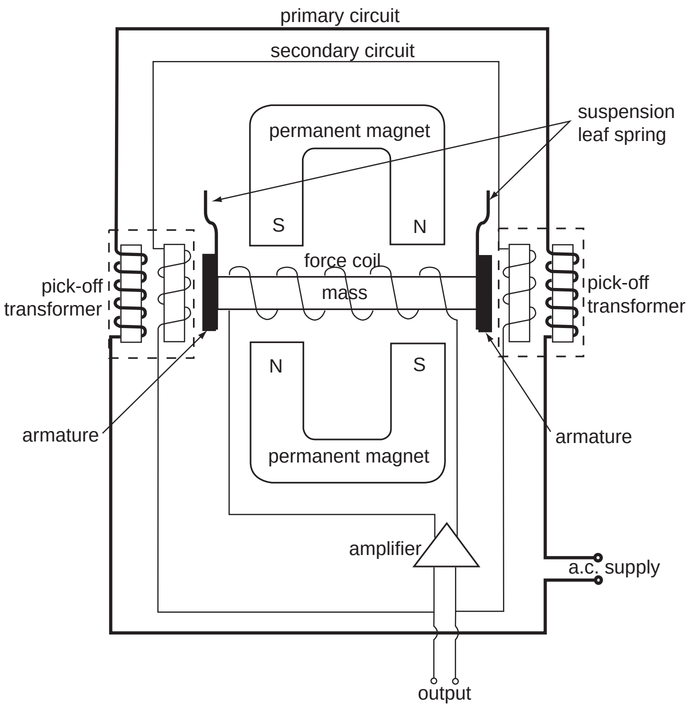

Figure 3.2 Accelerometer principle of operation.

When the accelerometer is subjected to an acceleration or deceleration along the axis of the pendulous mass, the mass is deflected by inertia, reducing the gap between one armature and its transformer and increasing the gap at the other end. This causes the voltages induced in the transformer secondary windings to be unbalanced, with a resultant current flow between them.

This current flow is amplified and fed back to the force coil surrounding the pendulous mass, creating an electro-magnetic field around the mass. This field reacts with the field produced by the permanent magnets on either side of the mass, to return the mass to its mid, or null, position. The

secondary current required to achieve this is directly proportional to the acceleration that caused the deflection and it therefore serves as the output signal from the accelerometer.

As previously stated, the purpose of the gyro-stabilised platform is to provide a mounting for the accelerometers that is earth horizontal at all times and that remains aligned with true north. The system uses rate integrating gyroscopes to sense platform horizontality and alignment. The platform is mounted on gimbals which are controlled by pick-off signals from the rate integrating gyros. Servo motors drive the gimbals to keep the platform level and aligned irrespective of aircraft maneouvres.

Rate integrating gyro

The rate integrating gyros typically used are single degree of freedom (i.e. single gimbal) gyroscopes that are highly accurate because the friction in the gimbal is reduced to virtually nil. The gyro rotor is mounted within a can-shaped gimbal, which is in turn mounted within a can-shaped case filled with viscous fluid. Transformers at either end of the cans create a magnetic field, upon which the inner gimbal is suspended, eliminating the need for mechanical bearings.

The gyro rotor is a two-phase synchronous motor rotating at about 24 000 rpm and, because it is a single degree of freedom gyro, it is sensitive to movement about one axis only, known as the input axis. Any movement about this axis will cause the gyro to precess, rotating the gimbal within which it is mounted. The gimbal is 'floating' within the outer casing and the relative movement between gimbal and casing is sensed by pick-off coils, generating an output signal. A schematic diagram of a rate integrating gyro is shown in Figure 3.3.

The gyro-stabilised (gimballed) platform

Three gyroscopes are mounted on the stable platform, fixed so that their sensitive axes are aligned with north–south, east–west and earth vertical, respectively. The platform is pivoted to inner and outer gimbals, as shown in Figure 3.4. Each gimbal axis is connected to a servo motor; the two horizontal-axis motors are known as the pitch and roll motors and the vertical axis motor is known as the azimuth motor.

The platform is shown in Figure 3.4 with the aircraft on a northerly heading and with the platform levelled (i.e. earth horizontal) and aligned with north. Under these circumstances the north gyro sensitive axis is aligned north–south and the east gyro sensitive axis is aligned east–west. Similarly, the north–south accelerometer pendulous mass axis is aligned north–south and the east–west accelerometer mass axis is

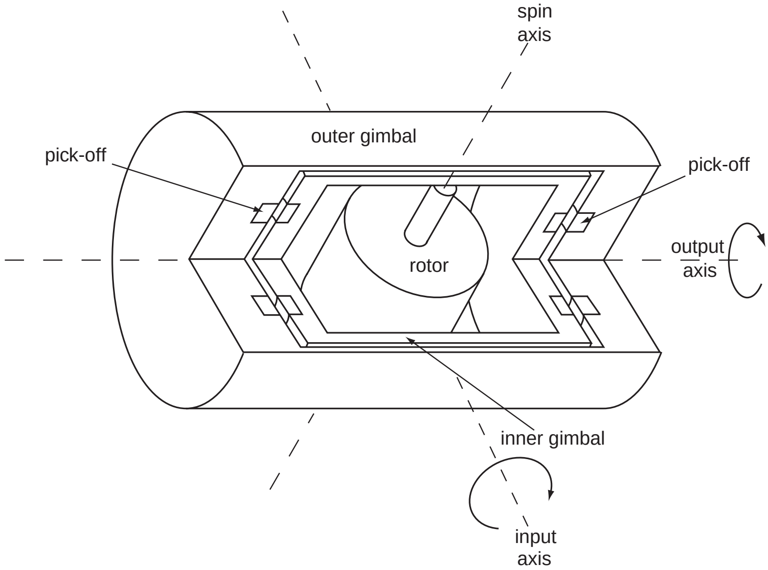

Figure 3.3 Rate integrating gyro schematic.

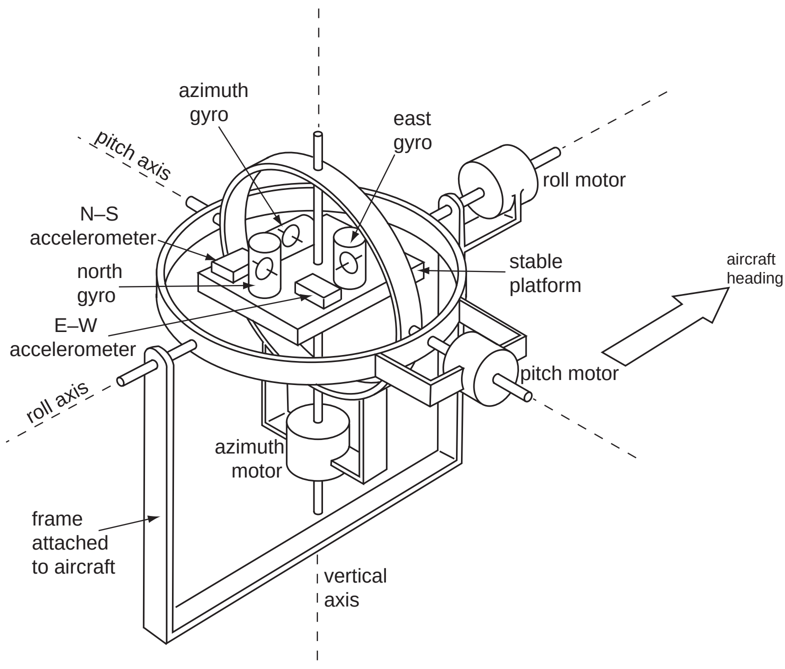

Figure 3.4 Gyro-stabilised platform – aircraft on northerly heading.

aligned east–west. The sensitive axis of the azimuth gyro is aligned with earth vertical.

Let us assume that the aircraft is stationary on the ground and accept for the moment that the system will maintain the platform both level and aligned. If the pilot now taxies the aircraft away from its stand, still on a northerly heading, the north–south accelerometer will sense the northerly acceleration as the aircraft moves away and the integrators will convert this to speed and distance for the computer and its cockpit display. The east–west accelerometer senses no acceleration and so the system continuously computes distance travelled north from the starting point.

On arrival at the eastern end of the east–west runway (very convenient airfield, this), the pilot turns the aircraft left onto a heading of 270°T. The frame of the stable platform, being attached to the airframe, will have turned through 90° with the aircraft, as illustrated in plan view in Figure 3.5.

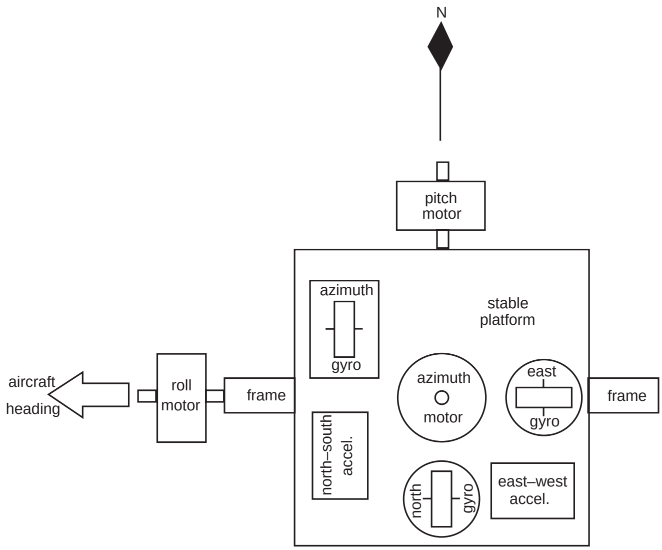

Figure 3.5 Gyro-stabilised platform – aircraft on westerly heading.

Any tendency of the stable platform to turn away from its north–south alignment will be immediately sensed by the azimuth gyro, which will precess and send an error signal to the azimuth servo motor. The platform is therefore turned relative to the airframe to maintain north–south alignment. The amount by which it is turned gives the change of aircraft heading for the INS computer.

As the aircraft accelerates along the east–west runway on the take-off roll, the acceleration is sensed by the east–west accelerometer and integrated to show increasing velocity and distance travelled westward. The aircraft nose

is pitched up and this nose-up pitch is maintained during the climb, which for simplicity we will assume continues on a westerly heading.

Any tendency of the stable platform to tilt in pitch will be immediately sensed by the north gyro, since on this heading the pitch axis is coincident with the north–south axis. The north gyro will send an error signal to the pitch servo motor to correct the tilt until the north gyro signal is nullified.

Finally in this example, let us assume that the aircraft levels out at the top of the climb at constant speed (i.e. no acceleration) and turns left onto a heading of 225°T. During the turn the aircraft will, of course, bank and any tilting of the stable platform will be sensed by both north and east gyros, which will signal the pitch and roll servo motors to maintain the platform level. North alignment during the turn is again maintained by the azimuth gyro and azimuth servo motor, turning the platform relative to the airframe and generating the heading change for the INS computer.

Whenever the aircraft is on a non-cardinal heading, control of platform levelling is shared by the north and east gyros and the pitch and roll servo motors. North–south alignment is at all times maintained by the azimuth gyro and its servo motor. The situation is illustrated in plan view in Figure 3.6.

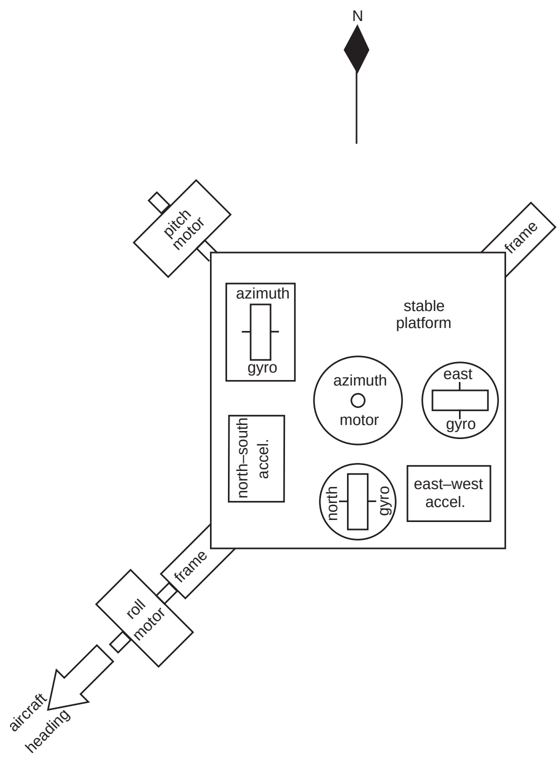

Figure 3.6 Gyro stabilised platform – aircraft on non-cardinal heading.

Position calculation

We saw in Figure 3.1 how two stages of integration convert acceleration into speed and distance travelled. The ultimate function of an inertial navigation system is to provide the pilot with navigational data such as track, groundspeed and present position in terms of latitude and longitude. The block diagram in Figure 3.7 illustrates how this is achieved.

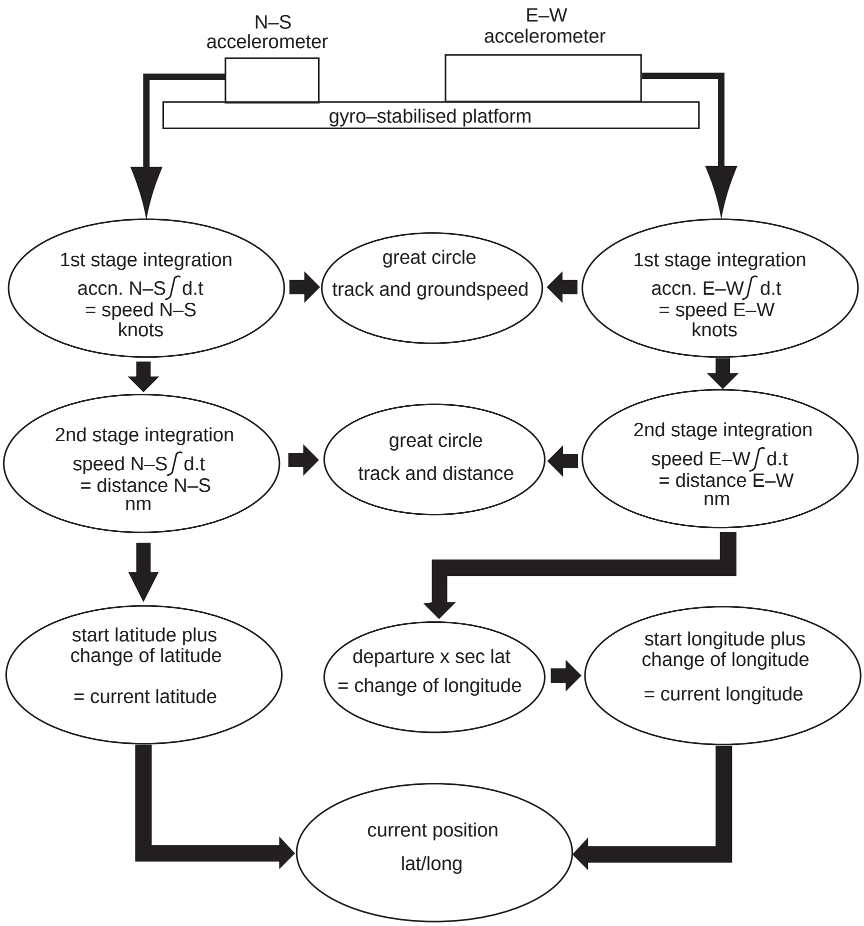

Figure 3.7 INS calculations.

Track and speed

The track calculated by the INS is a great circle track. The speed calculated at any instant is groundspeed, because the INS bases its calculations upon accelerations of the aircraft over the surface of the earth.

Latitude and longitude
Distance travelled north–south in nautical miles converts directly to change of latitude, since each nautical mile along a meridian equates to one minute of latitude. Distance travelled east–west is known as departure and is calculated using the equation:

$$\text{Departure E-W (nm)} = \text{change of longitude}' \times \text{cosine latitude}$$

Since, in the case of the INS computation, departure is known the calculation carried out by the INS computer to determine change of longitude is therefore:

$$\text{Change of longitude}' = \text{departure (nm)} \times \text{secant latitude}$$

INS self-alignment

The gyro-stabilising platform is self-levelling and self-aligning, but these functions can only normally be carried out in non-military aircraft with the aircraft stationary on the ground. In order to compute the speed, distance travelled and position of the aircraft the INS must first be referenced to north at the current aircraft position, which is fed into the computer by the pilot from airfield information. The process of self-alignment is performed in two distinct phases; first the platform must be levelled so that the accelerometers are not influenced by gravity and then it must be aligned with true north so that they only sense horizontal accelerations on the north–south and east–west axes.

Levelling and alignment are initiated by switching the INS to STANDBY and inserting the present position in terms of latitude and longitude, and then selecting ALIGN mode.

Levelling

Conventional systems reduce the time taken to level the platform by using gravity to achieve coarse levelling, followed by fine levelling using the accelerometers.

Coarse levelling
During coarse levelling the pitch and roll gimbals are typically driven by the servo motors until they are mutually perpendicular and the platform is brought to within about 1° of level using either gravity or reference to the airframe.

Fine levelling

With the aircraft stationary and the platform level there should be no output from the accelerometers. If there is an output from either or both accelerometers this indicates that the platform is not level and that acceleration due to gravity is being sensed. The output signals from the accelerometers are used to torque the north and east gyros, which in turn use the pitch and roll servo motors to drive the platform about the pitch and roll axes until output from the accelerometers is zero.

However, it will be appreciated that a platform levelled to earth horizontal using spatial referenced gyroscopes will not remain horizontal over an earth that is rotating. From Figure 3.8 it will be seen that a level platform above the poles would suffer drift about the vertical axis at earth rate (15.04°/hr), and one at the equator would suffer topple about the north–south axis at the same rate. At intermediate latitudes the platform would drift and topple at a rate dependent upon the latitude. To correct for this the north gyro is biased for topple at $15.04 \times \cos \text{lat}°/\text{hr}$ and the azimuth gyro is biased for drift at $15.04 \times \sin \text{lat}°/\text{hr}$.

Consequently, the platform will only be level when the output of the east–west accelerometer is zero and the output of the north–south accelerometer is also zero with the correct bias applied. The INS computer is able to apply the correct bias, because current latitude is inserted before the levelling procedure begins. Fine levelling typically brings the platform to within 6 seconds of arc of earth horizontal and takes about $1\frac{1}{2}$ minutes.

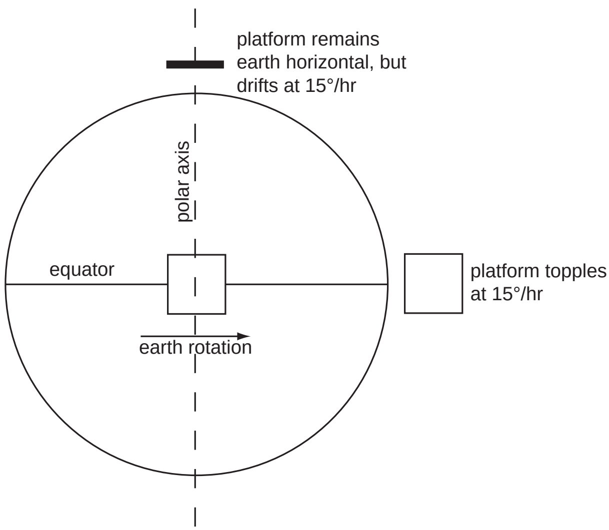

Figure 3.8 Effect of earth rotation on stable platform.

Gyro-compassing

The process of aligning the platform with the local meridian is usually referred to as gyro-compassing or azimuth alignment and it is the final stage of the alignment procedure. Most modern north-referenced gyro-stabilised platforms use a system called open loop gyro-compassing. Since both earth rate and the local latitude are known, any misalignment between the platform north–south axis and the local meridian can be calculated. When the inertial navigation system is switched from the alignment mode to the navigation mode the platform is rotated by the azimuth servo motor through the computed misalignment angle.

Many earlier systems use what is called closed loop gyro-compassing to achieve the same result. In this, any gravitational acceleration due to tilt of the platform sensed by the north accelerometer after fine levelling results in an output signal that is used to torque the azimuth gyro until the signal is nullified, which will only occur when the platform north–south axis is aligned with the local meridian. During this process the fine levelling process remains operative.

With either system the gyro-compassing phase takes between 6 and 10 minutes and typically achieves an alignment accuracy of about 6 minutes of arc. The time taken for self-alignment and the accuracy achieved increases with latitude. Close to the poles it becomes impossible to align a north-referenced platform; in UK latitudes self-alignment usually takes between 10 and 15 minutes.

Mode selector panel

The various modes of operation of the stable platform INS are selected by means of a simple panel, similar to that illustrated in Figure 3.9. The rotary switch shown has five positions and there are two illuminated annunciators.

STANDBY (SBY) is the mode selected to switch the system on. In this mode power is supplied to the system and the gyros are warmed and spun up to operating speed. Whilst this is taking place it is usual to insert the current position to the nearest 6 seconds of arc.

ALIGN selects the alignment mode, during which the levelling and alignment procedure described above takes place. When this is completed the **READY NAV** annunciator illuminates, indicating that the INS is ready for use.

NAV is the navigation mode, used throughout the period of ground movement and flight and during which the INS will make all its navigation

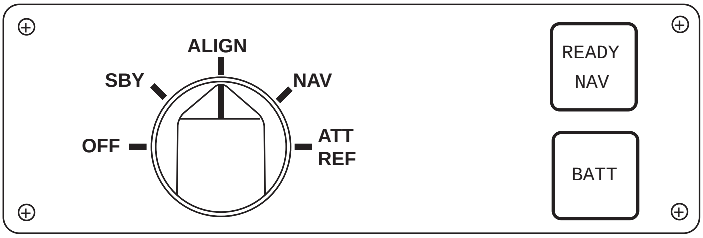

Figure 3.9 Mode selector panel.

calculations and display them as required on the control and display unit, to be described shortly.

ATT REF is the attitude reference mode and is only used when the INS computer fails to provide its navigational information. In this mode the stable platform is used to provide heading, pitch and roll information.

The INS has its own internal battery, which is capable of supplying power to the system for a limited period, typically about 15 minutes, in the event of loss of the normal power supply. The **BATT** annunciator will illuminate red when the battery power falls to a predetermined level, warning the pilot that the INS is about to fail.

INS error corrections

It has already been shown that the gyro-stabilised platform is subject to drift and topple due to earth rotation and that this is continuously corrected by computation of earth rate for the current latitude. In addition to this, the system is affected by transport wander and by coriolis effect.

Transport wander

As the platform is transported around the earth it will, if uncorrected, drift and topple. This is because it is spatially referenced and it is necessary for the INS computer to bias the rate-integrating gyros by an amount equal and opposite to the drift and topple rates in order to maintain the platform level and north-aligned. A biasing signal is applied to a torque motor on each gyro output axis, causing the gyros to send correcting signals to the servo motors.

Coriolis effect

Any vehicle moving over the face of the earth is subjected to coriolis effect. Put simply, coriolis effect is the combination of the vehicle movement and earth rotation and it produces acceleration to the right in the northern hemisphere and to the left in the southern hemisphere. As we know, any acceleration is sensed by the INS accelerometers and, if uncorrected, would

result in a false computation of track, groundspeed, etc. Given that the INS is continuously computing latitude, it is able to calculate and apply the necessary corrections.

The Schuler loop

The periodic rate of a pendulum, that is the rate at which it swings, is dependent upon the length of the pendulum. The longer it is, the longer it takes to complete a full cycle of swing from centre to left, to right and back to centre. Some considerable time ago a mathematician by the name of Schuler calculated that a pendulum the length of the earth's radius would have a periodic rate of 84.4 minutes. No doubt he had his reasons.

The relevance of this to the gyro stabilised platform is that it behaves as though it were attached to a pendulum from the centre of this earth, by virtue of the fact that it is maintained earth horizontal. Consequently, if the platform is subjected to a disturbance, such as shock or vibration, it is apt to oscillate with a frequency the same as a Schuler pendulum. The oscillations will, of course, move the platform away from earth horizontal and the accelerometers will produce an output that will be integrated into false speed and distance signals.

The oscillations will cause the platform to move from level to a maximum tilt (usually very small) over a period of 21.1 minutes, back to horizontal after 42.2 minutes, maximum tilt in the opposite direction after 63.3 minutes and back to horizontal again after 84.4 minutes. The effect on the speed integration, for example, might be an error of +3 knots at 21.1 minutes and −3 knots at 63.3 minutes. At 0, 42.2 and 84.4 minutes there would be no error and clearly the mean error over the full Schuler cycle is zero. This is known as a bounded error, since it does not increase with time. However, the positional calculations made from speed and distance will increase in error with time and are therefore known as unbounded errors.

Calculation of magnetic north

The INS computer is programmed with the values of magnetic variation at all geographic locations and can therefore apply this to the INS aligned true north to obtain magnetic north, so that track, heading, etc., may be displayed as either true or magnetic, as required.

Control and display unit (CDU)

A typical INS control and display unit panel is illustrated in Figure 3.10. The unit comprises an alphanumeric light-emitting diode (LED) display, a rotary selector switch and a keyboard for the insertion of data. In addition, there is

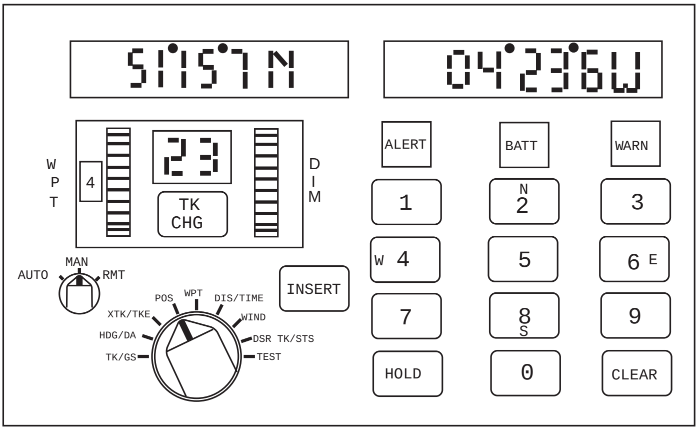

Figure 3.10 INS control and display unit.

the facility to set up a route by inserting waypoint latitudes and longitudes at which track changes are to be made. Waypoints are identified in numerical sequence using the selector wheel, starting with waypoint 1 as the departure airfield. Each waypoint is entered using the keyboard to set its latitude in the left alphanumeric display and its longitude in the right alphanumeric display. Insertion is made by pressing the INSERT button. If a change of track is authorised en-route, say direct from waypoint 5 to waypoint 8, this is made using the TK CHG pushbutton.

CDU selections

The displays associated with the various rotary display selector switch positions are listed and explained in the following paragraphs:

TK/GS (track and groundspeed)
The INS computed track, usually referenced to magnetic north, is displayed to the nearest tenth of a degree in the left display and the groundspeed in knots in the right display. For example, a current track of 135°M and a groundspeed of 467 knots would appear as 135.0° and 0467.

HDG/DA (heading and drift angle)
The heading obtained from the angle between the platform frame and north reference is displayed to the nearest tenth of a degree in the left display. The angular difference between heading and track (drift angle) is displayed to the nearest tenth of a degree in the right display, preceded by the letter R or L to indicate whether drift is right or left. Thus, a heading of 137°M on a track of 135°M would be presented as 137.0° and L 02.0°.

XTK/TKE (cross track distance and track error angle)
Cross track distance is the distance by which the aircraft is displaced right or left of the desired great circle track and is displayed in the left display to the nearest tenth of a nautical mile. The track error angle is the angular difference, right or left, between the desired great circle track and the actual track being made, to the nearest tenth of a degree. If the aircraft were displaced $1\frac{1}{2}$ nm to the left of the desired track of 135°M, the left display would read L 01.5. If the track being made good happened to be 130°M, the right display would read L 005.0°.

POS (present position)
The aircraft's current latitude and longitude are shown to the nearest 6 minutes of arc in the left and right displays, respectively. Suppose it happens to be at 51.15.7N, 04.23.6W, this would appear as 51°15.7′N in the left display and 04°23.6′W in the right display. This is the situation illustrated in Figure 3.10.

WPT (waypoint positions)
The position of each inserted waypoint is shown as latitude in the left display and longitude in the right display by selecting WPT on the rotary selector switch and scrolling through the waypoint numbers with the waypoint selector wheel.

DIS/TIME (distance and time to the next waypoint)
The distance in nautical miles from the present position to the next waypoint is shown in the left display and the time at present groundspeed to the nearest tenth of a minute in the right display.

WIND (wind speed)
The INS is able to compute wind direction and speed and these are displayed in the left and right displays, respectively, to the nearest degree of arc and knot.

DSR TK/STS (desired track and status)
The great circle track from one waypoint to the next changes as the aircraft progresses between the two and the INS computes the present desired magnetic track based upon distance from the waypoints, magnetic variation and the assumption that the aircraft is on track. This will appear in the left display to the nearest tenth of a degree and the right display will be blank. The status function is for use only whilst the INS is in ALIGN mode and it shows a numerical display in the right window that indicates the status of the alignment procedure. The display typically shows 99 at the start of

alignment and counts down to 0, when alignment is completed and READY NAV is illuminated.

Other CDU controls

It will be noted from Figure 3.10 that there are additional controls and annunciators and their functions are as follows:

The waypoint selection controls are situated immediately below the left LED display. A thumbwheel is rotated to select the number of a waypoint that is to be inserted or amended. An LED display indicates the current from/to situation; the illustration depicts the display that would appear when flying between waypoints 2 and 3. The track change (TK CHG) push button is used when altering the pre-planned sequence of waypoints. Conventionally waypoint 0 is the aircraft's current position. Suppose ATC has cleared you to fly direct to, say, waypoint 5 then the TK CHG button is pressed until 0–5 appears in the display. The DIM thumbwheel adjusts the brightness of the LED displays.

The AUTO/MAN/RMT rotary switch is used to select the type of flight control to be used in flying from waypoint to waypoint. In AUTO the INS will automatically change the from/to display as each waypoint is overflown and would normally be used in conjunction with automatic flight. In MAN (manual) the pilot is required to enter the from/to display as each waypoint is reached. The RMT (remote) position is used when two or more INS are fitted and enables the waypoint information to be transferred from one INS to the other(s).

The three annunciators situated below the right LED display serve to draw attention to specific events. The ALERT annunciator illuminates amber as the aircraft approaches the next waypoint, typically when about 2 minutes short of it, and will continue to flash until either cancelled by the pilot or, when in AUTO mode, by overflying the waypoint. The BATT (battery) annunciator will illuminate amber when the INS is operating on battery power, reminding the pilot that the system will only operate for a limited time on internal power. The WARN annunciator illuminates red in the event of a system failure. At the same time the status display will show a number that cross refers in the system manual to the nature of the failure.

Wander angle systems

It was mentioned earlier that the north referenced stable platform system becomes unacceptably inaccurate at high latitudes and is therefore useless for trans-polar flight. Because of this an alternative system has been developed in which the azimuth gyro is allowed to wander and the INS computer

transforms the wandering azimuth reference into any required earth co-ordinate.

Strapped-down systems

The disadvantages of the gyro stabilised platform INS are that the gyroscopes are expensive to manufacture and they inevitably suffer to some extent from random wander, however small, due to manufacturing imperfections. These, of course, lead to inaccuracies in the INS output. The gyroscopes take some time to warm up and reach their operating speed, and platform alignment is a relatively slow process.

With the vast improvements in computer technology and the introduction of the ring laser gyro it became possible to develop inertial systems that do not require a stabilised, earth horizontal platform, but which can be mounted directly to the aircraft structure. Hence the term 'strapped down'. These systems typically use three ring laser gyros with their sensitive axes aligned to the aircraft roll, pitch and yaw axes.

Since the accelerometers are also attached to the airframe and move with it, it follows that they will sense gravitational accelerations. The INS computer must differentiate between the accelerations that occur in the earth horizontal plane and those that occur in the aircraft horizontal plane in order to eliminate gravitational accelerations.

Alignment is achieved during an alignment phase with the aircraft stationary on the ground. The INS computer is able to discriminate between the accelerometer outputs due to gravity, since aircraft attitude is fixed, and those due to earth rotation and to compute the angle between the aircraft fore and aft (roll) axis and true north.

In the navigation mode the INS computer receives inputs of aircraft manoeuvres from the ring laser gyros and uses these to identify accelerometer outputs due to aircraft movement in the earth horizontal plane. These are then related to the north–south, east–west and azimuth axes to compute speed and direction of movement, distance travelled, current position, etc. Corrections for earth rate, transport wander and coriolis effect are computed in much the same way as previously described, from the calculated angular differences between aircraft and earth horizontal and between true north and the aircraft longitudinal axis.

Some strapped-down systems use an alternative type of ring laser gyro that has four sides as opposed to three and some military systems incorporate a third accelerometer to measure vertical acceleration.

Fibre optic gyro (FOG)

Some manufacturers employ fibre optic gyros (FOGs) instead of ring laser

gyros. In these the beam of light is carried in fibre optics instead of the much more expensive Cervit block required to transport the laser beam of the RLG.

Tuned rotor gyro

The tuned rotor gyro has a rotor that is, in its simplest form, flexibly suspended on a drive shaft so that it has limited freedom of movement relative to the shaft. In other words, the gyro rotor axis of rotation may be misaligned with the drive shaft axis by as much as 2°.

At a specific speed of rotation the centrifugal force of the spinning rotor, acting on the spring that flexibly attaches the rotor to the drive shaft, creates a 'tuned' condition in which the rotor behaves as though it were freely suspended, i.e. it behaves as a free gyro.

These gyroscopes are much less expensive to manufacture than the rate integrating gyro previously described and are generally more robust.

Sample questions

1. The output of the INS accelerometers is integrated . . . . . . with time to obtain . . . . and . . . . :

   a. Once distance velocity?
   b. Twice velocity distance?
   c. Twice distance velocity?
   d. Once velocity distance?

2. A typical gyro-stabilised platform contains:

   a. Two accelerometers and three rate-integrating gyros?
   b. Three accelerometers and two rate-integrating gyros?
   c. Two accelerometers and two rate-integrating gyros?
   d. Three accelerometers and three rate-integrating gyros?

3. With the aircraft on a northerly heading and the gyro-stabilised platform properly levelled and aligned, acceleration will be sensed by the:

   a. East gyro?
   b. North gyro?
   c. East–west accelerometer?
   d. North–south accelerometer?

4. Change of longitude in minutes is calculated by the INS computer by:

   a. Multiplying departure E–W by cosine latitude?
   b. Multiplying departure E–W by cosine longitude?

c. Multiplying departure E–W by secant latitude?
d. Multiplying departure E–W by sine latitude?

5. The ALIGN mode of the INS mode selector:

a. Is only used with the aircraft stationary on the ground?
b. Is used when inserting waypoint information?
c. May not be used until the READY NAV annunciator illuminates?
d. Is selected when the platform has levelled and aligned?

6. The effect of earth rotation on the stable platform is corrected by:

a. Sending a biasing signal to the accelerometers?
b. Sending biasing signals to the east–west accelerometer and the north gyro?
c. Sending biasing signals to the north gyro and the azimuth gyro?
d. Sending biasing signals to the east gyro and the azimuth gyro?

7. The process of aligning the stable platform with the local meridian is known as:

a. Coarse levelling?
b. Gyro compassing?
c. Fine levelling?
d. Attitude referencing?

8. The periodicity of a Schuler pendulum is:

a. 42.2 minutes?
b. 63.3 minutes?
c. 21.1 minutes?
d. 84.4 minutes?

9. An aircraft is displaced $2\frac{1}{2}$ nm to the right of its desired track of 265°M, on a heading of 272°M. With XTK/TKE selected on the INS CDU, the left and right displays would show:

a. R 02.5 R 007.0°?
b. R 02.5 L 007.0°?
c. L 02.5 L 007.0°?
d. L 02.5 R 007.0°?

10. A gyro stabilised platform INS that is not referenced to true north is known as:

a. A drift angle system?

b. A non-levelled system?
c. A wander angle system?
d. A non-aligned system?

11. A strapped-down INS typically uses:

a. Three ring laser gyros with their sensitive axes aligned north–south, east–west and earth vertical?
b. Three ring laser gyros with their sensitive axes aligned with the aircraft lateral, longitudinal and vertical axes?
c. Two ring laser gyros and two accelerometers?
d. Three rate-integrating gyros with their axes aligned north–south, east–west and aircraft vertical?

12. The advantage of a tuned rotor gyro over a rate integrating gyro is that:

a. Its speed of rotation is unimportant?
b. Its alignment is unimportant?
c. It has no moving parts?
d. It is less expensive to manufacture?

Chapter 4
Electronic Instrumentation

As the operation of transport aircraft and their systems has become increasingly automated and, at the same time, the number of flight deck crew members has been decreased, it has become impossible for the crew to monitor and control the automated processes using conventional instrument displays. To overcome this situation the traditional form of instrument has been almost entirely replaced by computer-generated displays projected upon a few cathode ray tube (CRT) screens. These screens usually combine the features of a number of the conventional instruments and, especially in the case of engine and system displays, normally only show the more essential information, with less critical information being selected by the pilots only as required.

The principal electronic display systems in use are the Electronic Flight Instrument System (EFIS) for the presentation and control of navigational information and, for presentation of engines and systems information, either the Engine Indicating and Crew Alerting System (EICAS) or Engine Centralised Aircraft Monitoring (ECAM). EICAS is generally used in aircraft of American manufacture and ECAM in Airbus Industries aircraft.

Electronic Flight Instrument Systems (EFIS)

The EFIS comprises two identical systems supplying the captain and first officer with navigational information on two display screens each, mounted one above the other. The upper display screen is an electronic attitude and direction indicator (EADI) and the lower screen display is an electronic horizontal situation indicator (EHSI). Each pilot's display has its own control panel, and a symbol generator from which the electronic representations on the screens are generated. A third symbol generator acts as a standby unit and may supply either of the pilot's displays in case of failure.

Each symbol generator receives inputs from all navigational sources, both internal and external, and interfaces between these inputs and the display screens to present the information in a standard format. In addition, the symbol generators perform the monitoring and control functions of the EFIS.

A block diagram of the EFIS process is shown in Figure 4.1.

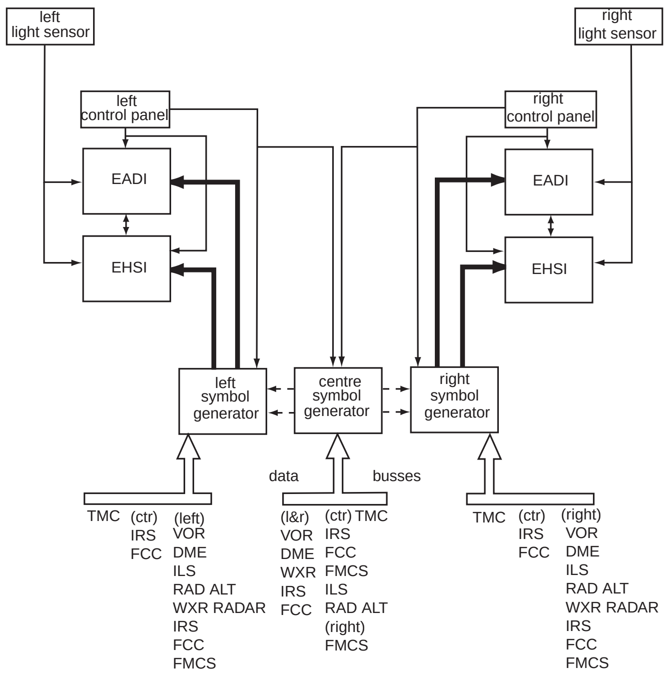

Figure 4.1 EFIS operating schematic.

It will be seen from Figure 4.1 that both the captain's and first officer's displays are provided with a control panel and a remote light sensor unit. The control panel is used to control the EADI and EHSI displays and the remote light sensor automatically adjust the brightness of the screen displays according to the light level on the flight deck. A typical EFIS control panel is illustrated in Figure 4.2.

The electronic attitude and direction indicator (EADI)

The EADI screen displays aircraft attitude in pitch and roll in the conventional format of an artificial horizon divided horizontally, with the upper half coloured blue and the lower half coloured yellow (or sand). The source data for the attitude indication are the aircraft inertial reference systems. The display also includes flight director command bars, ILS glideslope and localiser deviation indications, and deviation indication from a selected airspeed.

Radio altitude, decision height and operating modes of the automatic flight and autothrottle systems are also displayed on the EADI screen. Between 1000 ft and 2500 ft, radio altitude is displayed in digital format only, but below 1000 ft above ground level (agl) the display changes to include an

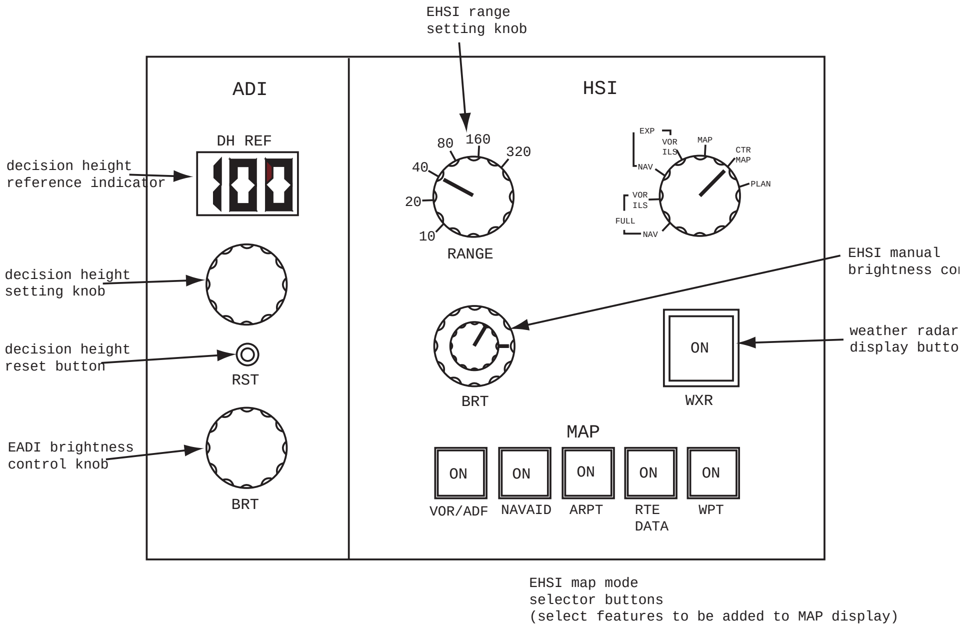

Figure 4.2 EFIS control panel.

analogue, circular scale display as well. As height agl decreases the white circular scale segments are progressively removed in an anti-clockwise direction. The decision height (DH) can be set by a control knob on the EFIS control panel and the selected DH is digitally displayed on the EADI screen and a magenta coloured marker appears at the selected height on the circular scale. At 50 ft above DH an aural chime begins to sound and its frequency increases until DH is reached. At this point the circular radio altitude scale and the DH marker both change colour to amber (yellow) and flash for several seconds. This alert can be cancelled by a push button on the EFIS control panel.

The display on both this and the EHSI screen is in colour and the colours used follow conventions laid down in JAR Ops 25 as listed below:

Display features should be colour coded as follows:

Warnings	Red
Flight envelopes and system limits	Red
Cautions, abnormal sources	Yellow/amber
Earth	Tan/brown
Sky	Cyan/blue
Engaged modes	Green
ILS deviation pointer	Magenta
Flight director bar	Magenta/green

Specified display features should be allocated colours from one of the following colour sets:

	Set 1	*Set 2*
Fixed reference symbols	White	Yellow
Current data, values	White	Green
Armed modes	White	Cyan
Selected data, values	Green	Cyan
Selected heading	Magenta	Cyan
Active route/flight plan	Magenta	White

The extensive use of yellow for other than caution/abnormal information is discouraged

In colour set 1, magenta is intended to be associated with those analogue parameters that constitute 'fly to' or 'keep centred' type information.

Precipitation and predicted turbulence areas should be colour coded as follows:

Precipitation (mm/hr)	0 to 1	Black
	1 to 4	Green
	4 to 12	Yellow/amber
	12 to 50	Red
	Above 50	Magenta
Turbulence		White or magenta

A typical EADI display screen is shown in Figure 4.3.

The electronic horizontal situation indicator (EHSI)

The lower of the two EFIS screens, the EHSI, presents a display of flight navigational information and progress in one of nine possible modes, selected from the HSI section of the EFIS control panel. The modes available are as follows:

- **MAP**. The display used for en-route navigational information and the one most commonly selected in cruise flight. The display shows features ahead of the aircraft, with the aircraft symbol appearing at the bottom of the display. This is illustrated in Figure 4.4.
- **CTR MAP**. Essentially the same as MAP, but the display is centred upon the aircraft current position, with an aircraft symbol in the centre of the display.
- **PLAN**. This display shows the planned route with waypoints and is principally used when entering waypoints into the flight management

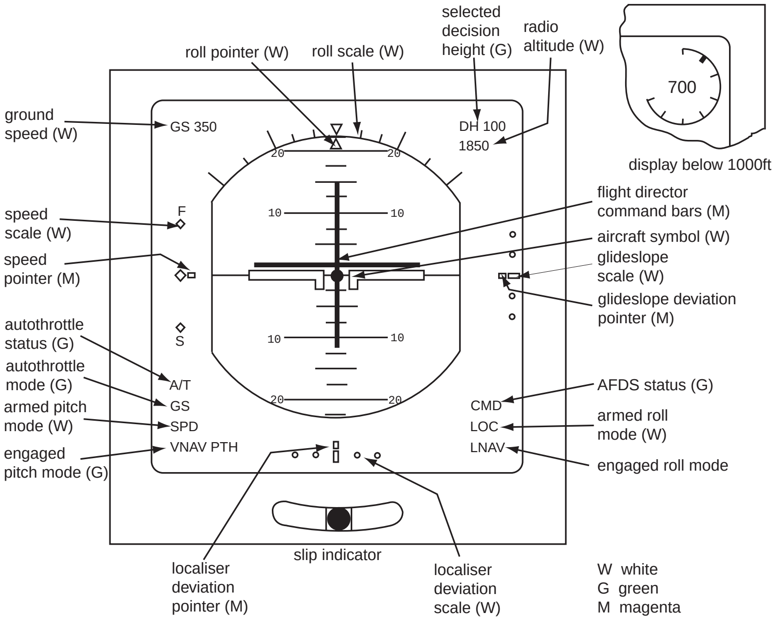

Figure 4.3 Typical EADI display.

system (FMS) computer before flight or when making changes to the planned route. It is illustrated in Figure 4.5.

- **FULL VOR/FULL ILS**. These displays are basically identical and show a compass rose with heading and deviation indications that follow conventional formats. They are used when checking aircraft track against a VOR bearing or ILS localiser. The display with ILS selected is shown in Figure 4.6. With VOR selected the display would be essentially the same, except that the in-use VOR would be indicated in the lower left corner, where ILS appears in the diagram.
- **EXP VOR/ILS**. In the expanded mode the information displayed is the same as in the full mode, but is in semi-map format. Only the relevant segment of the compass rose is displayed at the top of the screen, with a heading pointer. The aircraft's current (instantaneous) track is shown as a solid line extending from the aircraft symbol to the compass arc. The bearing of the selected radio aid, ILS or VOR, is shown as a solid line extending from the centre of the deviation scale to the compass arc. The display with ILS selected is shown in Figure 4.7. Again, the display with VOR selected is essentially the same. In either case the weather radar picture can be superimposed upon the display, if required.
- **EXP NAV/FULL NAV**. These two modes display lateral and vertical navigational information in much the same format as a conventional HSI.

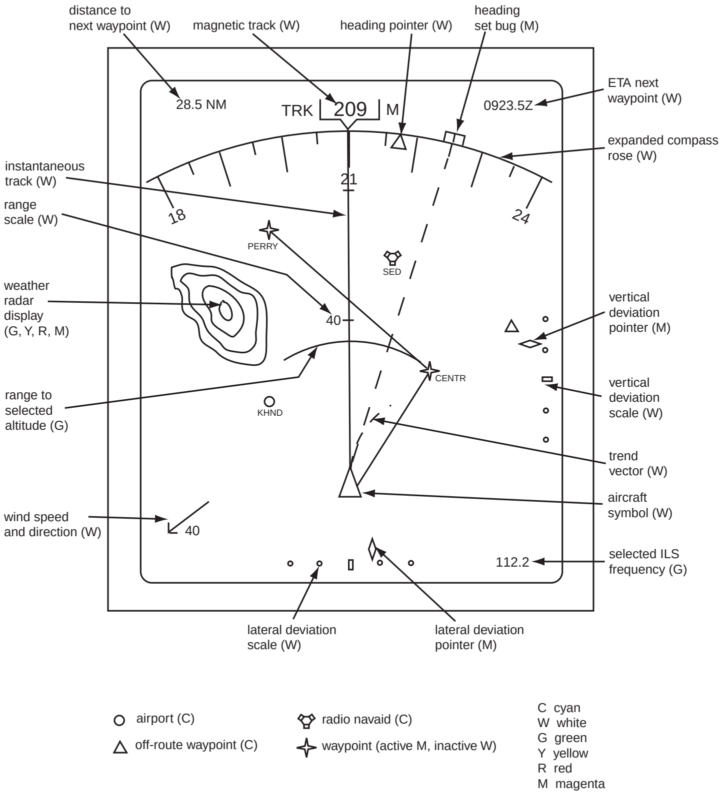

Figure 4.4 EHSI MAP mode display.

Expanded NAV mode shows a compass arc, whereas full NAV mode displays a full compass rose and does not permit the weather radar display to be superimposed, exactly as with the expanded and full VOR/ILS modes.

Map mode display

Between latitudes 65°S and 73°N the expanded compass rose may be referenced to either magnetic or true north as required, above those latitudes it may only be referenced to true north. Heading information is provided by the aircraft inertial reference systems; the heading and track pointers will, of course, only be aligned when there is no drift.

The vertical deviation scale and pointer indicates whether the aircraft is above or below the planned flight path and the lateral deviation scale and pointer whether it is to right or left of the planned flight path. Wind speed is indicated digitally in the lower left corner of the display, with an arrow indicating wind direction. The arrow is orientated to the map display, which

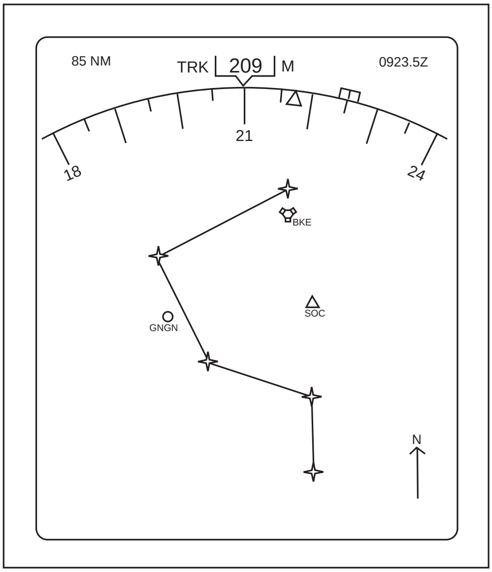

Figure 4.5 EHSI PLAN mode display.

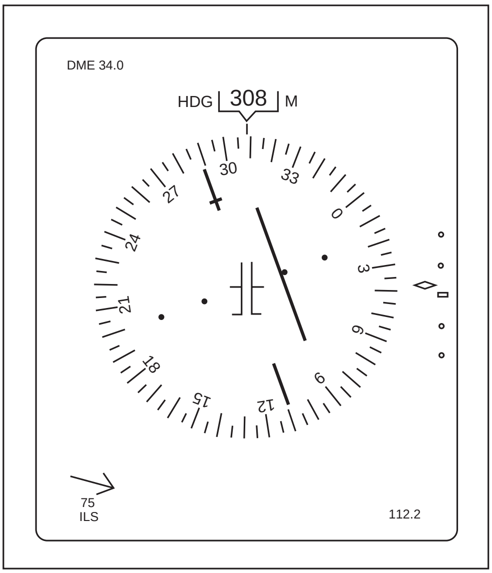

Figure 4.6 EHSI FULL ILS display.

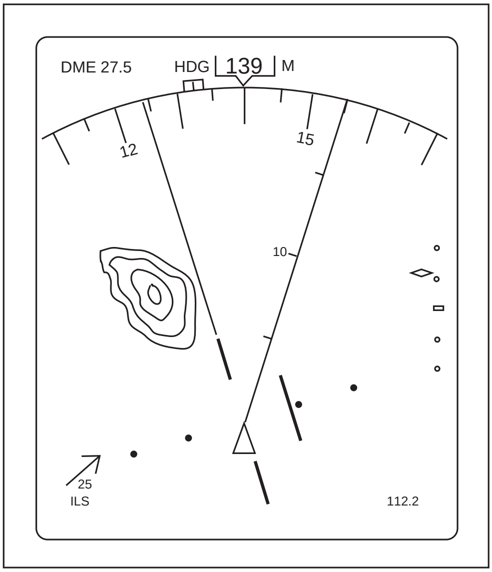

Figure 4.7 EHSI EXP ILS display.

is orientated to the aircraft track, so the wind direction is displayed relative to track.

The instantaneous track appears as a solid line extending from the apex of the triangular aircraft symbol to the compass rose. The selected range scale is superimposed and, when a planned change of altitude is taking place, an arc indicating range to the next altitude (at present rate of change of altitude) appears against this scale.

During change of heading a curved, dashed trend vector appears, showing predicted heading at the end of 30, 60 and 90 seconds from the present time. The planned flight path appears as a solid line from the apex of the aircraft symbol to the next, and subsequent, waypoints.

Navigational ground features, such as radio navigational aids and airports, are positioned on the display by the flight management system (FMS) from data obtained from VOR/DME stations and aircraft position is based upon these inputs together with those from the aircraft INS/IRS. When out of range of VOR/DME coverage the aircraft position is updated from INS/IRS only, but can be manually updated if required.

The weather radar display may be superimposed upon the MAP display, with intensity of radar returns indicated by green, yellow and red in order of intensity. Some displays will also indicate predicted turbulence, based upon very high intensity returns, in white or magenta.

PLAN mode display

The PLAN mode display, illustrated in Figure 4.5, is principally for use when entering or amending the lateral flight plan with the insertion of waypoints. The active route, from waypoint to waypoint, is displayed on the lower part of the screen, with significant navigational features such as airports and radio beacons included in their locations relative to the planned route. It should be noted that, in this mode, the display is orientated to true north. Wind speed and direction and the weather radar display cannot be superimposed whilst in PLAN mode. At the top of the display the expanded compass rose with track and heading as in the MAP display is maintained, together with distance and time to the next waypoint.

This display is particularly useful as a checking medium when inserting waypoint co-ordinates, before entering them into the flight management computer.

ILS mode displays

In ILS mode there are two possible screen displays, full and extended. These displays are primarily for use during landing approach. In the full ILS mode a complete compass rose fills the central part of the screen, with an aircraft symbol, deviation scale and deviation pointer indicating aircraft position relative to the ILS localiser beam superimposed, as illustrated in Figure 4.6.

The compass rose, driven by input from the inertial reference system, rotates against a fixed heading pointer as aircraft heading changes. The heading, magnetic or true, appears digitally within the pointer. Selected DME range is displayed in the top left corner of the screen, with wind speed and direction in the lower left corner. The selected type of radio beacon, in this case ILS, is also shown here, with the beacon frequency shown in the lower right corner. A vertical deviation scale and pointer, to indicate aircraft position relative to the glide path, is on the right side of the display.

With expanded (EXP) ILS mode selected, an expanded compass arc is displayed at the top of the screen, as in MAP and PLAN modes, with aircraft heading displayed in digital and analogue form as before. Selected DME range appears in the top left-hand corner of the screen, whilst wind speed and direction, and a reminder that the display is based upon ILS transmissions, appear in the lower left-hand corner. The selected ILS frequency appears in the lower right corner.

A semi-map display fills the lower part of the screen, with the triangular aircraft symbol positioned near the bottom of the display. A lateral deviation scale with deviation pointer intersects the apex of the aircraft symbol and a solid line extends from the central bar of the lateral deviation scale to the compass arc, indicating the bearing of the localiser transmitter. A second solid line, with a range scale, extends from the apex of the aircraft symbol to the compass arc, indicating the current (instantaneous) aircraft track. When

a new set heading is selected, a dotted line appears briefly from the apex of the aircraft symbol to the heading bug. In ILS mode a vertical deviation scale, with a pointer indicating aircraft position relative to the glide path, is on the right-hand side of the screen. Weather radar returns may be superimposed upon the display if required. The expanded ILS display is illustrated in Figure 4.7.

VOR mode displays

Full and expanded VOR mode displays are essentially the same as the ILS displays described above. These displays are useful when checking the aircraft track and heading relative to a selected VOR/DME and the principal differences in display features are that they will identify the selected beacon and indicate whether the aircraft is tracking toward (TO) or away from (FROM) the beacon. In VOR modes there is, of course, no vertical deviation indication.

Engine indicating and crew alerting system (EICAS)

The EICAS system is an electronic display consisting of two CRT screens mounted vertically, one above the other, and usually positioned centrally on the cockpit console, where they are easily visible to either pilot. The displays are capable of presenting all the engine and system operating data traditionally displayed by a mass of dials at a flight engineer's station, with facilities for displaying a great deal more information besides. The upper of the two screens, known as the primary display, normally shows only essential (i.e. primary) engine information such as engine pressure ratio (EPR), turbine spool speed (N_1) and exhaust gas temperature (EGT). The lower of the two screens, known as the secondary display, may be used to display less important (secondary) information and details of abnormal engine or system operating conditions.

The EICAS displays are generated by two computers that are continuously receiving operating data from the engines and the various aircraft systems. At any given time only one computer is operating the system, whilst the other functions as a standby. A display selection panel enables the pilots to select one of two operating modes, operational or status. A third mode, maintenance, is available on the ground, specifically for use by maintenance personnel. A block schematic diagram of a typical EICAS is shown in Figure 4.8.

Operational mode

This is the mode in which the system is used throughout flight. In this mode the upper screen displays the primary engine information listed above and the lower screen remains blank so long as all engine and system operating

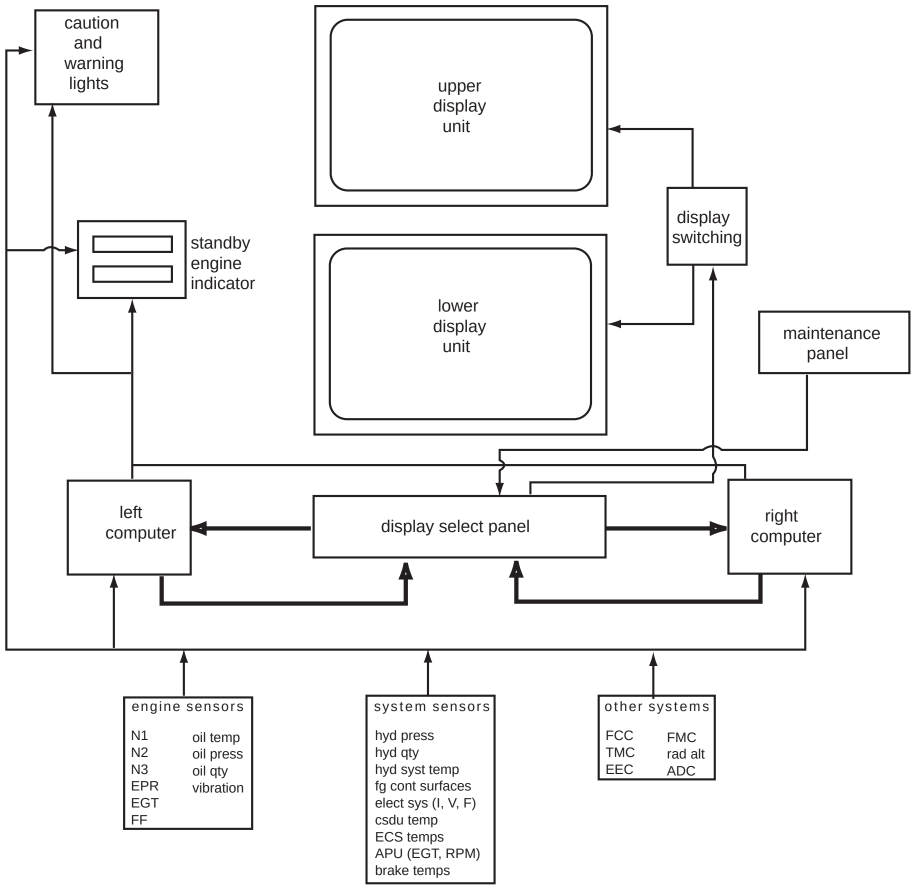

Figure 4.8 EICAS block schematic.

parameters are normal. In the event of an abnormal condition developing an alert message will appear on the upper screen and the lower screen will display details of the abnormal condition in analogue and digital format. Figure 4.9 shows the upper screen display with conditions normal.

The EICAS display is colour coded, following the JAR Ops 25 codes previously listed for EFIS displays.

When an abnormal engine or system operating condition develops, an appropriate warning or cautionary message will appear on the left side of the upper screen together with a row of pointers directing attention to the lower screen, where an analogue and digital display details the nature of the failure or critical condition. The alert messages relating to abnormal conditions are prioritised by the EICAS computer so that they appear in order of importance and degree of crew response required. Warning messages, requiring immediate corrective action, appear at the top of the screen in red.

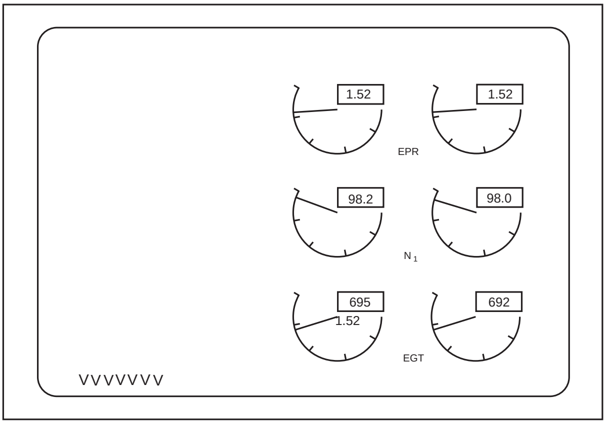

Figure 4.9 EICAS primary display.

Cautionary messages, requiring immediate crew awareness and possible remedial action, appear below the warning message(s) in yellow (amber). Both warning and cautionary messages are accompanied by an aural alert, such as a fire bell or a repeated tone. Advisory messages, which only require crew awareness, are indented on the display and also appear in amber. No aural alert accompanies these messages.

An EICAS display with primary alerts and secondary display is illustrated in Figure 4.10.

Status mode

This mode is primarily for use during preparation of the aircraft for flight and shows the status of aircraft systems and their readiness for flight. The information is allied to the aircraft minimum equipment list. The display appears on the lower screen of the EICAS and shows flying control surface positions in analogue format, with system status information in digital message format. The quantity of information available is too great for a single display and is available by selecting successive 'pages'. The number of the page being viewed is displayed on the screen. A typical status mode display is shown in Figure 4.11.

Maintenance mode

This mode is available to maintenance engineers for diagnosis of operating faults. It contains records of engine and system operating conditions and is only available with the aircraft on the ground. A separate control panel is provided for the display of maintenance data.

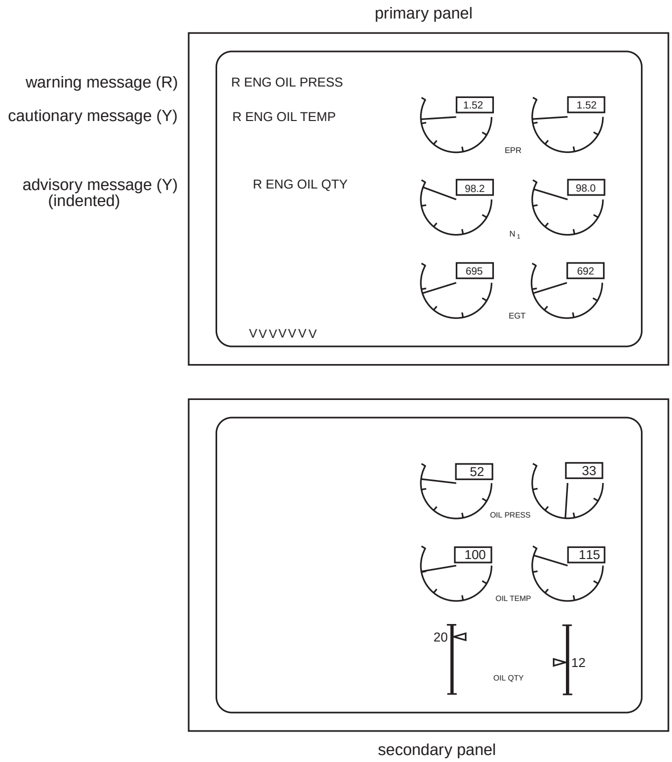

Figure 4.10 EICAS primary and secondary display.

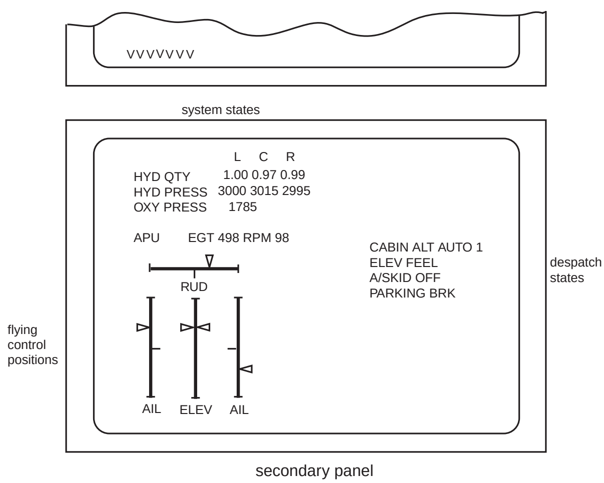

Figure 4.11 EICAS status mode display.

Display select panel

This panel, illustrated in Figure 4.12, is used to select the type of EICAS display required and is usually situated on the centre console adjacent to the display screens. The function of the various controls is as follows:

- **Display push buttons**. When the engine display push button is depressed secondary information appears on the lower screen. Depression of the status push button selects the status mode referred to above.
- **Event record push button**. Engine or system malfunctions in flight are recorded automatically and stored in the EICAS computer memory. Should the flight crew have reason to suspect that a transient fault has occurred, depression of the event record push button will highlight relevant data in the computer stored records for subsequent investigation by maintenance personnel. This latter is known as a manual event.
- **Computer rotary switch**. This is used to select the in-use computer of the system. In the AUTO position the left computer will normally be in use; switching to the right computer will occur automatically in the event of failure. The LEFT or RIGHT positions are used for manual selection of the in-use computer.
- **Brightness control**. This is a dual rotary switch. The inner knob controls display intensity and the outer knob controls the brightness balance between the two displays.
- **Thrust reference setting**. This is also a dual rotary switch. The outer knob is used to select an engine and the inner knob is pulled and rotated to position a cursor on the EPR or N_1 circular scale.
- **Maximum indicator reset**. If a measured parameter, such as engine oil temperature, exceeds a preset limit an alert will appear on the EICAS display. The maximum indicator reset push button is depressed to clear the alert when the excess condition has been rectified.

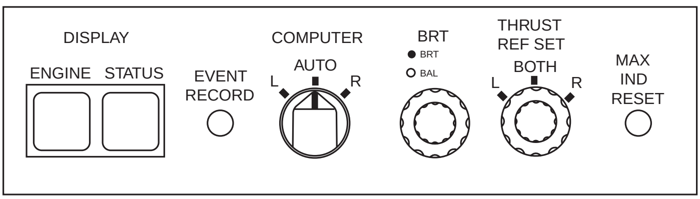

Figure 4.12 EICAS display select panel.

System failures

It has already been explained that, in the event of failure of one computer, the standby computer will take over either automatically or by pilot selection. If the lower display screen should fail whilst secondary information is

being displayed, the information is transferred to the lower half of the upper screen in abbreviated, digital format. This is known as a compact display and is illustrated in Figure 4.13. Failure of one display screen inhibits use of status mode. Should both display screens fail, a standby engine indicator displays essential engine performance data in a liquid crystal diode (LCD) display, as illustrated in Figure 4.14. The indicator has a two-position control switch. With this switch in the AUTO position the standby indicator is functioning, but does not display any data unless the CRT displays are not functioning. With the switch in the ON position the unit displays continuously. The test switch has three positions and is used to test the alternative power supplies to the indicator.

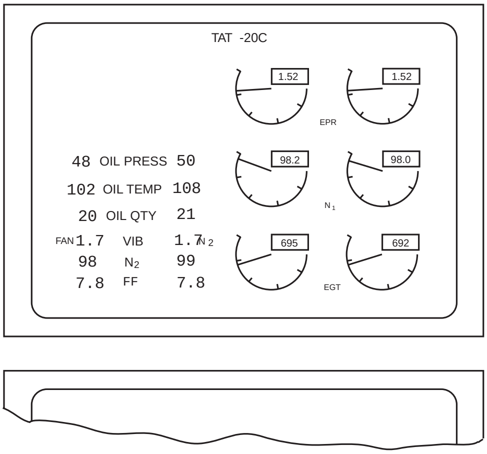

Figure 4.13 EICAS compact display.

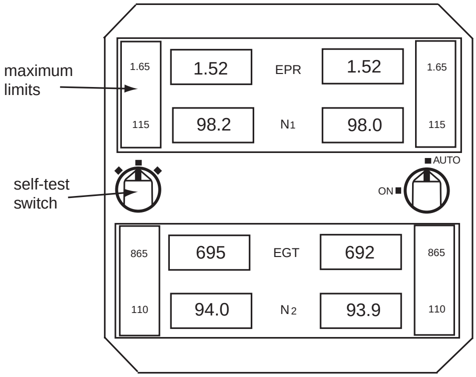

Figure 4.14 Standby engine indicator.

Failure of the display select panel is indicated on the upper EICAS screen, which continues to display primary engine information. Secondary information still automatically appears on the lower screen, but the panel control switches are inoperative.

Electronic centralised aircraft monitoring (ECAM)

The ECAM system was developed for the Airbus A310 aircraft and the block schematic diagram in Figure 4.15 illustrates that version of the system. It is principally an aircraft systems display medium, with primary engine information displayed on traditional instruments. The ECAM display screens are mounted side-by-side and both are in use continuously. The left screen displays information covering systems status, warnings and corrective actions required in check list format. The right screen shows associated information in analogue displays.

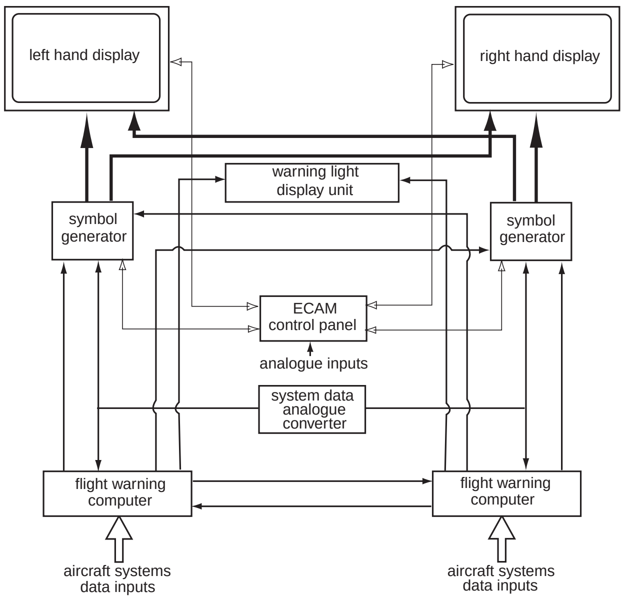

Figure 4.15 ECAM block schematic diagram.

Control panel

The ECAM control panel is illustrated in Figure 4.16. The left and right display control knobs are for switching on the displays and adjusting the display brightness. The functions of the various push button switches are as follows:

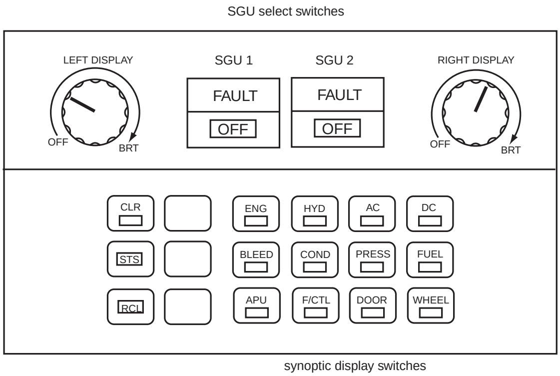

Figure 4.16 ECAM control panel.

- **SGU select**. In normal operation of the system both symbol generator units (SGUs) are functional. In the event of a fault being detected by an SGU self-test circuitry, a fault caption is illuminated on the appropriate switch. Releasing the switch isolates the affected SGU and extinguishes the fault caption, illuminating the OFF caption in its place.
- **Clear (CLR)**. This is a clear switch, which will illuminate whenever a warning or status message appears on the left screen. Depressing the switch clears the message.
- **Status (STS)**. Depressing this switch allows manual selection of aircraft system status displays, provided that there is no warning message displayed.
- **Recall (RCL)**. If a warning message is cleared whilst its associated failure condition is still existent, it may be recalled by depressing the RCL push button.
- **Synoptic displays**. Synoptic diagrams of each of the 12 aircraft systems are called up on the right screen, provided that there is no warning message displayed, by depressing the appropriate synoptic display switch.

Operating modes

The system has four operating modes, known as NORMAL, ADVISORY, FAILURE and MANUAL. Apart from the last-named, the display modes are automatically selected.

- **NORMAL mode**. This mode is flight-related and is the mode in which the system normally operates throughout the flight progress from pre-flight through to post-flight checks. In this mode the left screen displays system states in check list format and the right screen contains a relevant pictorial

display. For example, during post engine start checks the left screen would typically list the state of each of the aircraft's systems and the right screen would display the various system states pictorially (e.g. hydraulic, electrical, pressurisation, etc.). These displays are selected by depressing the appropriate push button on the ECAM control panel, illustrated in Figure 4.16.

- **ADVISORY mode**. The display automatically switches to this mode when the status of a system changes. For example, starting the APU will cause a message to that effect to appear on the left screen. The right screen will continue to display the selected diagram.
- **FAILURE mode**. This mode takes precedence over all others and is automatically selected by the ECAM system in the event of normal operating parameters being exceeded in any of the aircraft systems. An appropriate warning message appears on the left screen, accompanied by an aural alert. Below this message, the corrective actions required by the flight deck crew are listed. On the right screen a diagrammatic display of the affected system illustrates the fault. When the corrective action has been taken the displays change to illustrate this. Examples of these displays are shown in Figures 4.17 and 4.18. In the example shown the No. 2 generator frequency is outside permitted limits and disconnection of the constant speed drive is called for, with the situation

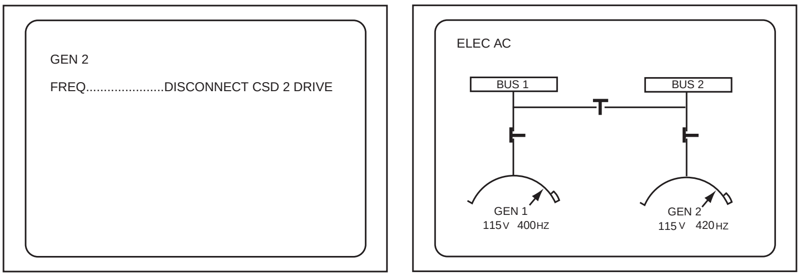

Figure 4.17 FAILURE mode display before corrective action.

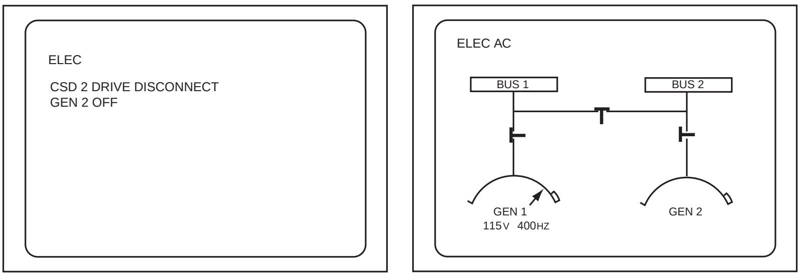

Figure 4.18 FAILURE mode display after corrective action.

displayed pictorially on the right screen. When the corrective action has been taken, the left screen display changes to show that No. 2 generator is disconnected and the right screen display shows the new status of the a.c. electrical system.

- **MANUAL mode**. Provided that there are no warning messages displayed on the left screen, diagrams related to the aircraft systems can be called up on the right screen by depressing any of the 12 synoptic push buttons on the control panel.

Sample questions

1. The EFIS display comprises:

   a. Two screens mounted side-by-side on a central console?
   b. A pair of screens mounted one above the other in front of each pilot?
   c. An upper screen showing the horizontal situation and a lower screen showing attitude and direction?
   d. A left screen showing navigational information and a right screen showing engine and systems displays?

2. Symbols coloured magenta on the EFIS display indicate:

   a. Warning information?
   b. Command information?
   c. Cautionary information?
   d. Current situation information?

3. Below 1000 feet agl, the radio altitude display on the EADI is:

   a. A digital only display?
   b. An analogue display of a pointer moving around a circular scale?
   c. An analogue and digital display with the selected decision height displayed in digital format only?
   d. An analogue display consisting of a circular scale in which the segments disappear as height decreases, with height agl displayed digitally?

4. At 50 ft above decision height:

   a. An aural chime alert sounds and increases in frequency until DH is reached, at which point the circular scale and DH marker both change colour to amber and flash for several seconds?
   b. An aural chime alert sounds and increases in frequency until DH is reached, at which point the circular scale and DH marker both change colour to red and flash continuously until cancelled?

c. An aural chime alert sounds and continues until cancelled. At DH the circular scale disappears and the DH marker changes colour to amber?
d. An aural chime alert sounds for several seconds. When DH is reached, the circular scale and DH marker both change colour to magenta and flash until touchdown?

5. In MAP mode, a curved dashed line extending from the apex of the aircraft symbol indicates:

a. Rate of change of aircraft heading?
b. Range to selected altitude?
c. Predicted heading at the end of 30, 60 and 90 seconds from preset time?
d. Deviation from desired track?

6. Weather radar returns are available in the following EFIS EHSI modes (answer a, b, c or d):

1. MAP
2. PLAN
3. EXP ILS
4. FULL ILS
5. FULL VOR
6. EXP VOR

a. 1, 2, 3, 4, 5, 6?
b. 1, 2, 3?
c. 1, 3, 6?
d. 1, 2, 4, 6?

7. The expanded compass rose in EFIS EHSI modes may be referenced to:

a. Magnetic or true north between latitudes 65°S and 73°N?
b. True north only above latitudes 63°N or S?
c. Magnetic or true north between latitudes 63°S and 75°N?
d. Magnetic or true north between latitudes 75°S and 75°N?

8. In PLAN mode the planned route display:

a. Is orientated to the present aircraft track?
b. Is orientated to true north?
c. Is orientated to the present aircraft heading?
d. Is orientated to magnetic north?

9. A circle coloured cyan on the EFIS MAP display indicates:
   a. An off-route waypoint?
   b. An active waypoint?
   c. An airport?
   d. A radio beacon?

10. In the engine indicating and crew alerting system:
    a. There are two computers, both of which are operating at all times?
    b. There are two computers, but both will only operate simultaneously if so selected manually?
    c. There are three computers, two in operation and one on standby at any time?
    d. There are two computers, normally one is operating and the other is on standby?

11. EICAS has:
    a. Three modes, all of which are available during flight?
    b. Three modes, of which only two are available during flight?
    c. Four modes, all of which are available during flight?
    d. Four modes, three of which are available during flight and the fourth on the ground only?

12. EICAS advisory messages:
    a. Appear on the primary display in analogue form, coloured yellow?
    b. Appear on the secondary display in digital and analogue form?
    c. Appear on the primary display in digital form, indented and coloured red?
    d. Appear on the primary display in digital form, indented and coloured yellow?

13. In the event of failure of an EICAS display screen:
    a. Primary and secondary information will be displayed on the remaining screen in compact form?
    b. Primary engine information only will appear on the standby engine indicator?
    c. Primary and limited secondary information will appear on the standby engine indicator?
    d. Primary engine information only will appear on the remaining CRT screen?

14. Failure of the EICAS display select panel will:

    a. Render the entire system inoperative?
    b. Reduce the display to the standby engine indicator only?
    c. Not affect the primary and secondary displays, but will be indicated on the primary display?
    d. Reduce the display to a compact version on one screen only?

15. The ECAM system has four operating modes:

    a. The mode principally used during flight is known as the operational mode?
    b. The manual mode may only be selected on the ground?
    c. The advisory mode may only be selected during flight?
    d. The failure mode takes precedence over all other modes?

16. Warning messages on CRT displays are required to be coloured:

    a. Red?
    b. Yellow?
    c. Cyan?
    d. Magenta?

17. An armed mode on an EFIS EADI display will appear as:

    a. A magenta symbol?
    b. A white analogue feature?
    c. A white alphanumeric message?
    d. A cyan digital message?

18. Predicted turbulence will appear on the EHSI weather radar display as an area of:

    a. Red or cyan?
    b. Red or magenta?
    c. White or magenta?
    d. Red or white?

Chapter 5
Automatic Flight Control

Automatic control of an aircraft in flight has developed from relatively simple autopilot systems, in which the aircraft was automatically maintained on a steady heading, to complex systems that automatically control all aspects of aircraft flight in terms of lateral and vertical navigation (LNAV and VNAV) and speed from immediately post take-off to the end of the landing roll and beyond.

To achieve this, the autopilot requires inputs from all navigational sources, both internal and external to the aircraft, and the engine thrust must be managed at all flight stages for optimum power at optimum economy. The co-ordination of all these requirements is achieved in modern transport aircraft by the flight management system (FMS).

Flight management system

Almost all modern passenger transport aircraft employ a computerised flight management system, the purpose of which is to reduce crew workload, thereby enabling a reduction in crew numbers, and to achieve the best possible fuel economy with the overall result that operating costs are minimised.

The system may function purely as an advisory unit, providing the flight crew with advice on the control settings required to achieve optimum fuel economy in each of the various flight conditions such as take-off, climb, cruise, descent and approach. In most cases this is done by displaying the necessary engine pressure ratio (EPR) or torque settings and the recommended climb/descent rates. In this type of flight management system all the control adjustments are made manually by the crew in response to the FMS advisory messages. Associated with this type of FMS will be a discrete flight director system, providing the flight crew with command and advisory information regarding the vertical and lateral flight path of the aircraft through an attitude director indicator (ADI) and horizontal situation indicator (HSI). The flight director system will be described later in this chapter.

More sophisticated flight management systems interface with the aircraft automatic flight control and automatic thrust control systems to achieve fully automatic control of all flight phases with the exception of take-off and

initial climb-out. In these systems the management of the aircraft planned flight path is divided into lateral and vertical profiles. The FMS guides the navigation of the aircraft vertically (VNAV) to achieve the planned altitude at each stage (waypoint) of the planned flight and laterally (LNAV) to arrive overhead each geographical turning point (waypoint) of the planned flight. At each stage of the flight the FMS instructs the automatic thrust control system as to the power setting necessary to achieve maximum fuel economy. A typical flight management system profile is shown in Figure 5.1.

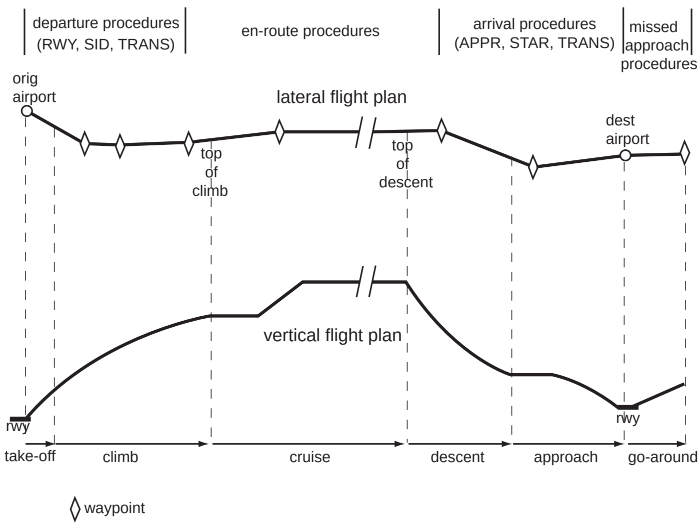

Figure 5.1 Typical FMS flight profile.

In addition to the control functions performed by the FMS, it continuously provides information to the flight deck displays such as EFIS, EICAS or ECAM, as described in the preceding chapter. Flight director commands to the flight crew, particularly necessary during take-off and initial climb-out, are transmitted through the EADI and EHSI functions of the EFIS, as we have already seen to some extent.

In order to perform its multitudinous functions, the FMS must be provided with navigational data from all the navigation systems such as INS/IRS, DME, VOR and Doppler and from all the engine and associated systems monitoring equipment.

Since the term IRS has appeared in this text, it is perhaps appropriate at this stage to differentiate between an inertial navigation system (INS) and an inertial reference system (IRS). The former, as described in Chapter 3, is a stand-alone navigation system that does not interface with other systems. The IRS is essentially the same insofar as it performs basically the same

navigational functions, but its computer system is considerably more sophisticated to permit interfacing with the flight management system.

Figure 5.2 shows a block schematic diagram of the data inputs and outputs of a typical flight management system.

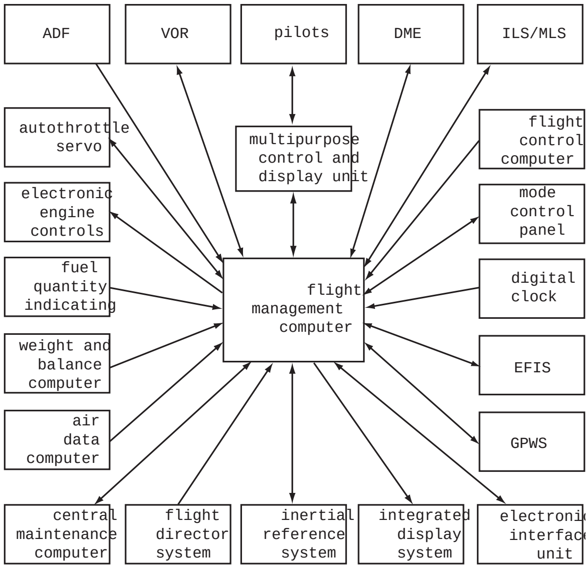

Figure 5.2 FMS data interfacing.

Multi-purpose control and display unit (MCDU)

The link between the flight crew and the FMS is the multi-purpose control and display unit. It provides the crew with the means to make inputs to the system in order to obtain required displays, or information to assist with decision-making in respect of the aircraft's flight progress. The CDU display is in paged, digital format on a small, typically 2 × 3 inch, CRT screen.

The MCDU is primarily used for long-term actions, such as monitoring and revising the flight plan, selection of operating mode and insertion of data such as aircraft weight, wind speed and direction, various temperatures and performance data. It provides the flight management computer (FMC) with readout capability, together with verification of the data entered into the computer memory. Flight plan and advisory data are continuously available for display on the MCDU.

The MCDU panel has a full alpha-numeric keyboard, along with mode, function and data entry keys. The keyboard includes advisory annunciators, display light sensors and a manual brightness control. A typical multi-purpose control and display unit is illustrated in Figure 5.3.

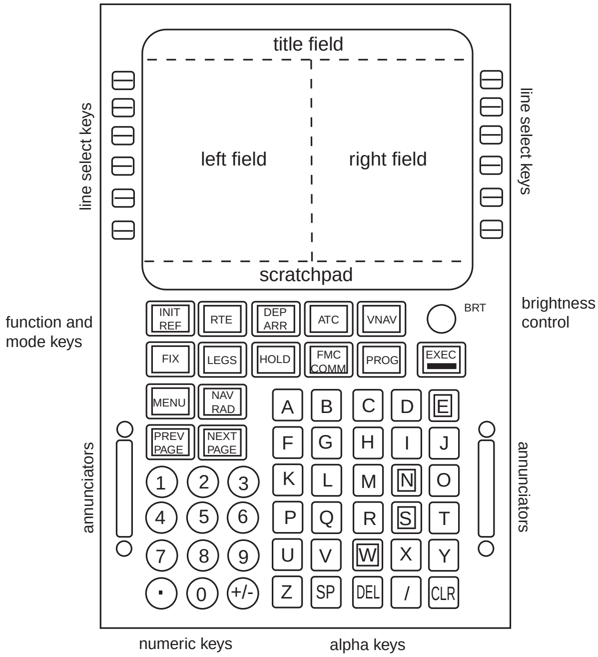

Figure 5.3 Typical MCDU.

Display screen

The display screen of the MCDU shown in Figure 5.3 has 14 lines with a total of 24 characters per line. The page format of the screen is divided into four areas, these are:

- Title field, which contains the title of the page of subject data displayed and the page number (e.g.1/2, meaning page 1 of 2).
- Left and right fields, each containing six pairs of lines of 11 characters per line. The pilot has access to one line of each pair through the line select keys on either side of the unit.
- Scratchpad, which forms the bottom line of the display screen. Scratchpad entries may be pilot-inserted, unless an FMC originated message is displayed in this field, and they are independent of the page displayed.

Line select keys

Momentarily depressing a line select key affects the line adjacent to the key on the respective side of the MCDU for entry, selection or deletion of data.

Brightness control

The light intensity of the MCDU display may be adjusted by rotating the BRT knob. Brightness of the illuminated keys is automatically adjusted by a remote flight deck control.

Annunciators

There are two annunciators on each side of the keyboard. These display the following messages when appropriate:

- **DSPY**. Display (upper left). The white display light illuminates when the active lateral or vertical leg performance mode is not displayed on the current MCDU page.
- **FAIL**. Fail (lower left). The amber light illuminates when there is a fault in the FMC.
- **MSG**. Message (upper right). The white light illuminates to indicate to the pilot that an FMC-generated message is displayed on the scratchpad, or is waiting to be displayed when the scratchpad is cleared.
- **OFST**. Offset (lower right). The white light is illuminated when lateral navigation (LNAV) is based on a route parallel to, but offset from, the active route.

Alpha-numeric keys

These keys allow the pilots to enter letters and numbers onto the scratchpad successively from left to right. They include space (SP), delete (DEL), slash (/), and plus/minus (+/ −) keys.

Function keys

The function keys control the MCDU field displays, accomplished by the execution of pilot inputs and requests. The purpose of each of the function keys is briefly described below:

- **EXEC**. The execute key is the command key for the FMCS. Whenever a modification or activation is pending, a white light bar illuminates. Depressing the key will activate the flight plan, change the active flight plan or change the vertical profile, as appropriate.
- **NEXT PAGE**. Depressing this key causes the display to page on to the next higher page number in multi-page displays.
- **PREV PAGE**. Depressing this key causes the display to page back to the next lower page number in multi-page displays.
- **CLR**. The clear key extinguishes the MSG annunciator light and clears any message from the scratchpad. Where more than one message is displayed, each momentary push clears a single message; multiple messages are cleared by repetitive momentary pushes or a single long push.
- **DEL**. Pressing this key inserts the word DELETE onto the scratchpad, provided that the pad is clear. Line selection by means of a line selection key deletes the entered data on that line, but is only available for specific pages.

Mode keys

The mode keys control the type of page displayed on the MCDU and are therefore the means by which the pilots gain operational access to the flight management system. There are twelve keys, as described below:

- **INIT REF**. (Initialisation/reference.) The initialisation/reference key selects the first of a series of pages used to initialise the position of the FMCS and the inertial reference system (IRS).
- **RTE**. (Route.) The route key provides access to planned routes and selects the page for entering or changing the point of departure, destination or route.
- **DEP ARR**. (Departure/arrival.) Depressing this key calls up an index listing all terminal area procedures.
- **ATC**. (Air traffic control.) This key selects the ATC automatic dependent surveillance status page.
- **VNAV**. (Vertical navigation.) Depressing this key provides access to the climb (CLB), cruise (CRZ) and descent (DES) pages for evaluation and modification.
- **FIX**. The fix key provides access to the fix information pages, which are used for creation of waypoint fixes.
- **LEGS**. The legs key provides a page for evaluating or modifying lateral or vertical details of each route leg.
- **HOLD**. The hold key calls up the page for entering, exiting or amending a holding pattern.
- **FMC COMM**. (Flight management computer communications.) In most current systems this key is non-operational.
- **PROG**. (Progress.) This key is used to select current dynamic flight and navigation data, such as ETAs and fuel remaining at a given point (e.g. next two waypoints, destination or alternate).
- **MENU**. The menu key provides access to other aircraft subsystems and to the alternate control for the EFIS and EICAS control panels in the event of failure.
- **NAV RAD**. (Navigational radio.) Depressing this key selects the page for monitoring or modifying navigational radio tuning.

Figure 5.4 shows a typical MCDU page display.

The MCDU is duplicated so that each pilot has access to the system and the two units are usually located on either side of the central console. Operation of the FMS is fully described in the Aircraft Operating Manual for each aircraft type.

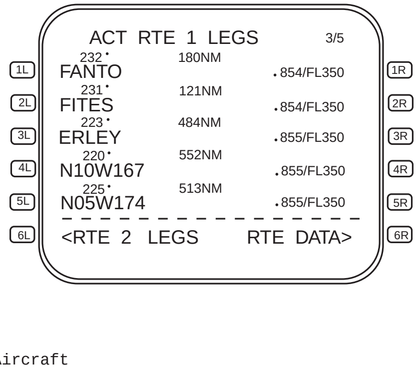

Figure 5.4 Typical MCDU display.

Flight director systems

A flight director system (FDS) integrates the display of aircraft attitude, in terms of roll and pitch, with radio navigational data to provide a complete directional command function for both vertical and lateral navigation. The components of a typical FDS are shown in block schematic form in Figure 5.5. The vertical gyro unit referred to in the diagram is similar to the attitude indicator described in Chapter 2, but instead of providing attitude information directly to a display it transmits attitude-related electronic signals to the flight director computer. In aircraft not equipped with a flight management system and EFIS the outputs from the computer are fed to an attitude director indicator (ADI) and a horizontal situation indicator (HSI).

Attitude director indicator (ADI)

The attitude director indicator display closely resembles that of the gyro attitude indicator, but with vertical and lateral deviation indicators and a radio altitude readout added. A fixed aircraft symbol, typically in the form of a flattened triangle, is positioned centrally against a background tape that is able to scroll up or down to indicate aircraft pitch attitude. The tape is coloured to represent sky and ground, with a horizontal dividing line representing the horizon. The tape is driven by a servomotor, which receives signals from the pitch channel of the vertical gyro unit.

The background tape and its roller drive are mounted upon a ring gear and this is rotated by a second servomotor that receives signals from the roll channel of the vertical gyro unit, to indicate roll attitude. The tape has $\pm 90^\circ$ freedom of movement in pitch and 360° in roll. Bank angle is indicated by a pointer attached to the ring gear, which moves against a fixed scale on the instrument casing.

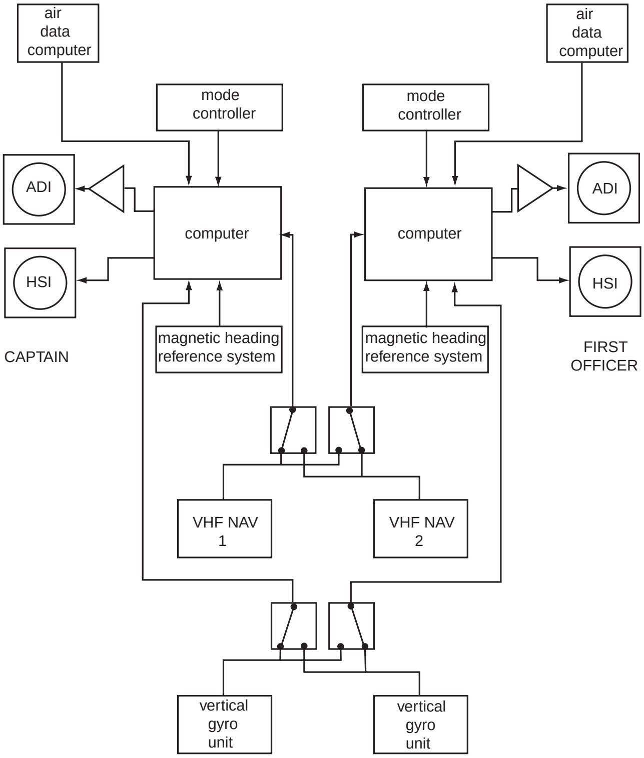

Figure 5.5 Flight director system block diagram.

Deviation from the ILS localiser beam is indicated by a pointer and lateral deviation scale at the bottom of the display. Below this is a conventional ‘ball and tube’ slip and skid indicator. Deviation from the glideslope is indicated at the left-hand side of the display by a pointer and vertical deviation scale. A radio altitude readout of height agl is displayed, typically during the last 200 ft of descent. This may take the form of a scrolling digital readout, as in Figure 5.6, or a ‘rising runway’ directly beneath the fixed aircraft symbol.

It will be noted that the ADI includes command bars. These are driven in response to signals from the flight director computer and they indicate pitch and roll commands to the pilot. The two bars are not physically connected to each other, but they normally move as a pair and the aircraft must then be flown to position the triangular aircraft symbol in the ‘vee’ formed by the bars, in order to satisfy the command. The principle is illustrated in Figures 5.7 and 5.8.

In Figure 5.7 the command bars have moved to demand ‘fly up’ and ‘fly left’. In order to satisfy the flight director command the pilot must therefore pitch the aircraft up into a climb and bank it to the left. Figure 5.8 illustrates

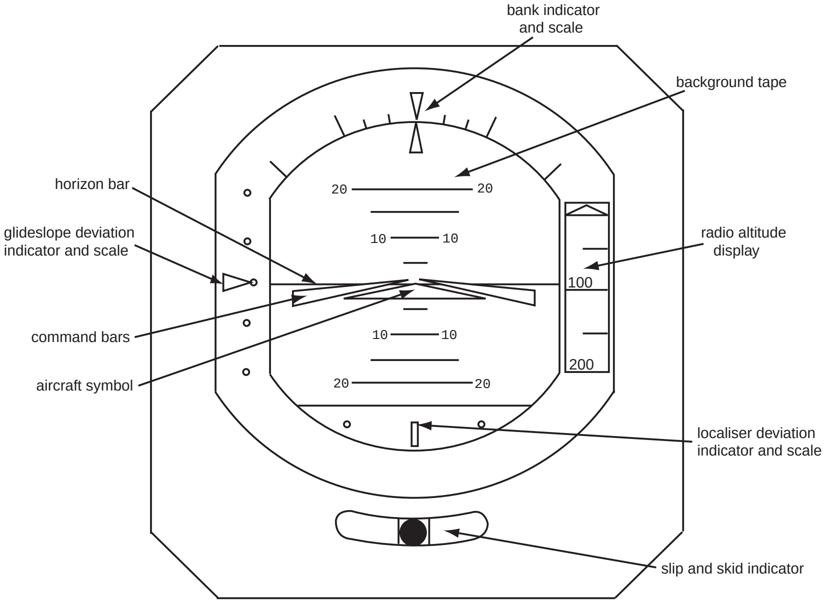

Figure 5.6 Attitude director indicator.

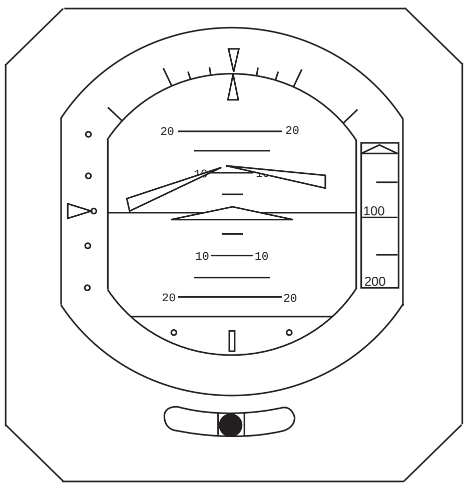

Figure 5.7 Pitch and roll commands unsatisfied.

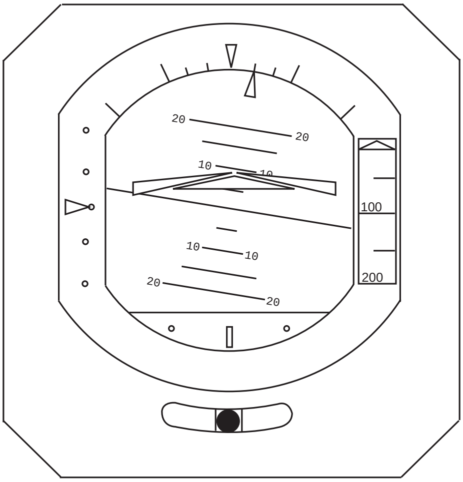

Figure 5.8 Pitch and roll commands satisfied.

the ADI display when these manoeuvres have been carried out and the aircraft is satisfying the climbing left turn command.

Some ADIs are equipped with a pitch command facility with which the pilot can select a given climb or descent in certain modes of operation of the flight director system. This will position the command bars to indicate the required pitch attitude.

Horizontal situation indicator (HSI)

The HSI presents a display of the lateral aircraft situation against a compass rose. The compass rose is driven by a servomotor receiving signals from the aircraft magnetic heading reference system. In aircraft fitted with an inertial navigation system the servomotor may also receive signals from this, enabling either magnetic or true heading to be selected. Aircraft heading is indicated by a fixed lubber line on the instrument casing.

The aircraft symbol, in the form of a miniature aircraft, is fixed at the centre of the display and points toward the aircraft heading lubber line. Control knobs at the bottom of the display allow a VOR/localiser course and a desired heading to be selected. Rotation of the course selection knob rotates the course arrow to indicate the selected course on the compass rose. At the same time, a digital course counter at the upper right of the display provides a readout of the selected course. The centre section of the course arrow is a movable lateral deviation bar, which is deflected left or right of the course arrow to indicate deviation from the selected VOR radial or localiser centre line. It thus indicates fly left or right to intercept the localiser beam or VOR

radial and the command bars of the ADI will move to direct roll in the appropriate direction. A to/from arrow indicates whether the selected course is to or from the VOR. The dots of the deviation scale represent displacement of approximately $1\frac{1}{4}^\circ$ and $2\frac{1}{2}^\circ$ from the localiser centre line or approximately 5° and 10° from the VOR radial. Once a course is set, the course arrow rotates with the compass rose as aircraft heading changes. In VOR mode of operation the deviation bar begins to indicate when the aircraft reaches approximately 16° from the radial; in ILS mode, movement of the bar begins at approximately 4° from the localiser beam centre.

The heading select knob, at the lower left side of the display, is used to set a desired heading. When it is rotated, a triangular heading bug moves against the compass scale to indicate the heading selected. In the heading mode of operation the ADI command bars will move to direct roll in the appropriate direction until the desired heading is attained.

Vertical deviation from the selected ILS glideslope is indicated by a pointer moving against a deviation scale. Range to a selected DME is displayed digitally at the upper left side of the HSI.

Figure 5.9 illustrates a typical HSI display.

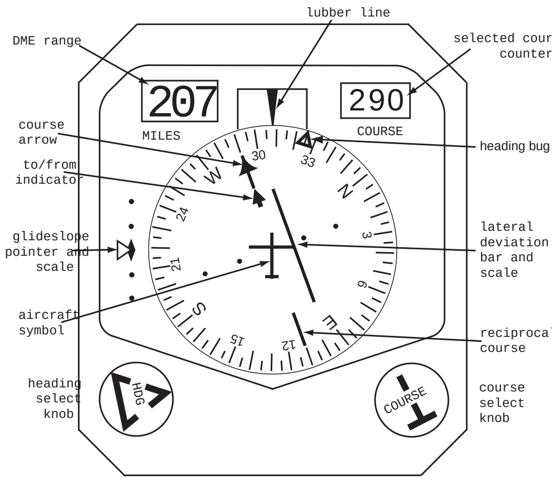

Figure 5.9 Horizontal situation indicator.

Warning indications

In the event of a fault developing it is vital that the pilot should be aware that the display may be in error. The command signals generated by the flight director system are continuously monitored and, should they weaken to the point that the information provided is unreliable, small red warning flags appear at the relevant part of the ADI or HSI display.

The ADI has three warning flags to indicate failure of the ILS glideslope receiver, the vertical gyro and the computer. These are labelled GS, GYRO and COMPUTER, respectively. When the GS flag is activated it obscures the glideslope pointer and deviation scale; the GYRO and COMPUTER flags appear at the bottom of the tape display.

The HSI also has three warning flags, labelled GS, COMPASS and VOR/LOC. The GS flag operates in the same manner as that on the ADI, whilst the COMPASS flag is activated in the event of failure of signal from the magnetic heading reference system (MHRS) and the VOR/LOC flag indicates failure of the VOR or localiser signal.

Modes of operation

The flight director system may be operated in a number of different modes and these vary with the various system manufacturers and aircraft types. Similarly, the method of mode selection by the pilots varies between systems, but is usually achieved through push button selector switches.

The basic operating modes common to most systems are described briefly below:

- **OFF**. With the system switched off the command bars of the ADI are retracted from view and the indicator is used purely for attitude reference.
- **HDG**. With heading selected, the ADI command bars provide guidance in roll to reach and maintain the compass heading selected with the heading select control knob and indicated by the heading bug. Where a pitch command facility is fitted this will be enabled in this mode.
- **VOR/NAV**. The ADI command bars provide guidance in roll in order to capture and maintain a selected VOR radial or ILS localiser beam and lateral deviation is indicated on both ADI and HSI displays. Pitch command facility is also enabled in this mode.
- **GS**. With glideslope selected the ADI command bars provide vertical and lateral guidance to capture and maintain the ILS glideslope and localiser beams. Lateral and vertical deviation indicators are activated on both displays.
- **GS AUTO**. This mode is basically the same as GS mode, except that interception and capture of the glideslope is automatic once the localiser beam has been captured.
- **ALT**. In this mode the ADI command bars provide guidance in pitch to maintain a preselected altitude.
- **APPR I**. This mode is used when making an approach to a Category I ILS. Lateral and vertical guidance for the capture and tracking of glideslope and localiser beams is provided by the ADI command bars.

- **APPR II**. Guidance is the same as for APPR I, but beam tracking is of a higher standard to meet the requirements of Category II ILS.
- **GA**. (Go-around.) After a missed approach using one of the approach modes, this mode may be selected for the execution of a go-around procedure. The ADI command bars will order a wings level, pitch-up attitude. Once the required power and speed settings have been achieved, HDG and IAS modes may be selected.
- **IAS**. This mode is used when it is necessary to maintain a given airspeed during climb-out or descent. The ADI command bars provide guidance in pitch.
- **VS**. This mode is selected when guidance is required for a specific rate of climb or descent (vertical speed). The ADI command bars provide guidance in pitch.
- **MACH**. This mode is for use at higher altitudes. Guidance is the same as for the IAS mode.

Figure 5.10 shows an example of a mode selector panel.

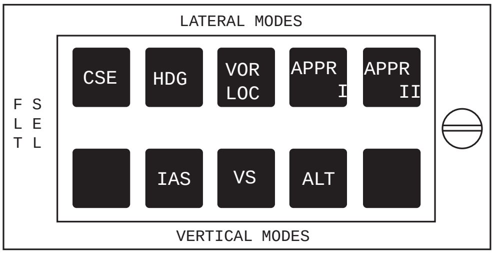

Figure 5.10 Flight director mode selector panel.

Gain programme in approach mode

The system response is optimised during an ILS approach by means of a programmer in the gain control section of the FD computer. Following capture of the localiser and glideslope beams the pitch and roll signals and the deviation signals are reduced to allow for the convergence of the beams during descent. The programmer is selected automatically at predetermined positions on the localiser and glideslope.

Lateral and vertical beam sensors

The task of these sensors is to supply input data to the FD computer to assist with the task of stabilising and adjusting the aircraft attitude as necessary. Disturbances about the aircraft's lateral and longitudinal axes are sensed

and signalled to the FD computer, which generates pitch and roll command bar movements in response. The commands are computed to reflect the rate of change of deviation due to disturbance.

Automatic flight control systems

The core of every automatic flight system is the autopilot. This is an autostabilisation system capable of maintaining the aircraft in stable flight about one or more of the aircraft's three axes of roll, pitch and yaw. Early autopilot systems were only designed to maintain wings level flight by controlling the aircraft in roll, that is about its longitudinal axis, and such a system is still in use today in some light aircraft. This is known as a single axis, or single channel, autopilot providing lateral stabilisation.

Inner loop control

The single axis lateral stabilising autopilot employs a closed loop control system in which the aircraft attitude in roll is controlled by operating the ailerons through a servomotor, or actuator. Any change in roll attitude is sensed by a rate gyro sensitive only to movement about the aircraft longitudinal axis. Such movement will cause the gyro to precess and this precession will be picked off and transmitted as an error signal to a controller, which compares input signals with the 'wings level' message programmed in its memory. The error signal will indicate the rate and direction of deviation from wings level and the controller will generate a correcting output signal of corresponding amplitude and rate of change. This signal is amplified and transmitted to the servomotor, which moves the ailerons in the appropriate direction to arrest the roll and restore the aircraft to wings level. The restoring movement is sensed by the roll gyro, with the result that the error signal received by the controller diminishes as the wings approach the level condition, until there is no error when the wings are once again level, by which time the servomotor has returned the ailerons to their neutral position. A simple block diagram of such a closed loop automatic control system is shown in Figure 5.11.

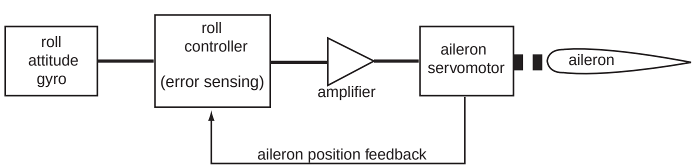

Figure 5.11 Single-axis autopilot system.

It will be appreciated that a system such as that described is only capable of maintaining the wings level, because the only information the controller has is the programmed desired condition, the actual condition as sensed by the rate gyroscope and response feedback from the aileron servomotor. It is incapable of maintaining a particular heading or course, since there is no directional information input to the system. Such a control loop is known as an inner loop, since it has no external references. The 'wings level' reference programmed into the controller may be, and almost invariably is, adjustable by the pilot. By rotating a knob, for example, on a control panel the desired roll attitude can be biased to left or right and the controller will signal the servomotor to move the ailerons and roll the aircraft until the required bank angle is achieved. The autopilot loop will now maintain that bank angle until the reference attitude is re-adjusted. Thus the pilot can control aircraft heading through the autopilot system.

A second autopilot inner loop channel can be added, to provide automatic control of the aircraft about the pitch, or lateral, axis. Once again, this is purely an autostabilisation system that will maintain a preselected pitch attitude. The reference pitch attitude may be adjusted by the pilot, as before, in order that the autopilot maintains a selected aircraft attitude for level flight, climb or descent.

An autopilot system that has both roll and pitch channels is known as a two-axis autopilot and this is probably the most common form of autostabilisation system. A third, yaw channel is often provided in larger transport aircraft, especially where stability about the yaw axis is a problem, as is the case with many swept-wing aircraft.

Outer loop control

Few transport aircraft fitted with autopilots use them just for autostabilisation. Instead, the autopilot is supplied with inputs from external sources, which direct it to control the aircraft laterally to maintain a preselected heading or course and to control it longitudinally to maintain a given altitude or vertical speed. Once this is done, it is a relatively straightforward progression to supply inputs, via the flight management system, from the aircraft's navigational systems to direct the autopilot so that it flies the aircraft along a preselected lateral and vertical flight path. These external control inputs are known as outer loop controls and a block diagram illustrating one channel of an autopilot system with inner and outer loops is shown in Figure 5.12.

Listed below are some of the sources of outer loop control signals supplied to the autopilot systems of a typical modern large transport aircraft:

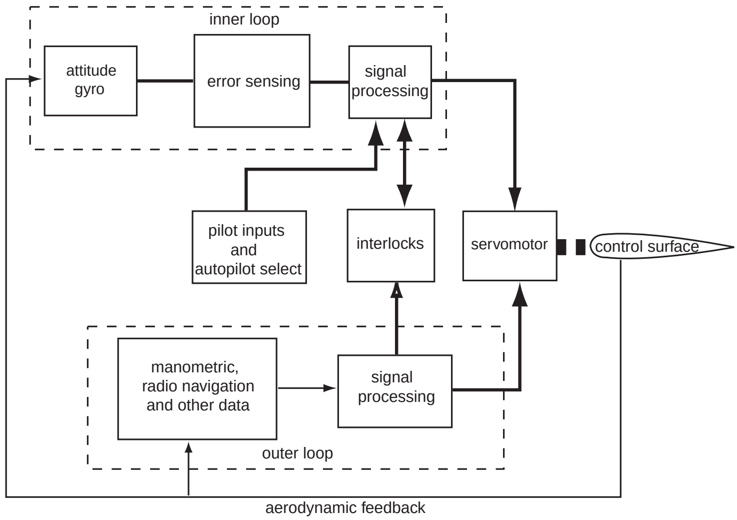

Figure 5.12 Autopilot inner and outer control loops.

- **Central air data computer (CADC)**. Manometric data of airspeed, pressure altitude, vertical speed and mach number are supplied by the CADC for control of the aircraft in pitch to select and maintain a given airspeed (airspeed select and hold), select and maintain a given altitude (altitude select and hold) or maintain a given vertical speed or mach number (V/S or mach hold). These are all pitch modes.
- **Magnetic heading reference system**. The aircraft magnetic heading is selected and maintained by control about the roll axis, using inputs from the MHRS. This is a roll mode, known as heading select and hold.
- **Radio navigation aids**. The ADF, VOR and ILS localiser receivers provide signals for lateral guidance of the aircraft. In these modes of operation the autopilot uses control about the roll axis to achieve a required course and they are consequently roll modes. The ILS glideslope receiver provides signals for vertical navigation, through control about the pitch axis, during approach and landing phases. This is, of course, a pitch mode of operation.
- **Flight management system**. The lateral and vertical information programmed into the flight management computer, when supplied to the autopilot systems, enables them to navigate the aircraft vertically and laterally along a predetermined flight path. Through this source many refinements are possible, such as pitch trim to compensate for fuel consumption to name but one. In this mode of operation (LNAV and VNAV) the FMS will be coupled to the roll, pitch and (where fitted) yaw channels of the autopilot.

Control surface actuation

Autopilot operation of the aircraft flying controls may be by hydraulic power-operated actuators, as is usually the case in large aircraft, or by servomotors otherwise known as servo-actuators. These are electrically driven mechanical, hydraulic or pneumatic machines that move the control surfaces by linear or rotary motion. In most cases the same servomotors operate the control surfaces whether the signal controlling them is from the autopilot or from the human pilot's controls.

In some aircraft control systems the servomotors are connected in series with the pilot's controls. In this case, when the autopilot is engaged and the flying control surfaces are moved, there is no corresponding movement of the pilot's controls on the flight deck. Alternatively, the servomotors may be connected in parallel with the pilot's controls and autopilot-initiated movement of the control surfaces will be mirrored by the flight deck controls.

Torque limiters

The stress loads imparted to the airframe when the control surfaces of a large aircraft are deflected can be enormous. Consequently, large or rapid deflections can conceivably induce structural loads beyond the design limitations of the aircraft, resulting in serious structural damage. In manual control of flight this danger is largely averted by artificial 'feel' devices, which warn the pilot if excessive control deflection is attempted. Since the autopilot system has no such 'feel' it is necessary to introduce safeguards into the control surface operation to prevent overloading of the airframe.

Torque limiting devices are inserted in the drive between servomotor and control surface, which will either slip or disengage if the torque required to achieve the attempted rate of deflection exceeds a preset limit. This device has the added benefit of preventing a servomotor runaway (undemanded operation of the servomotor) from more than slight deflection of the associated control surface, before disengagement occurs. The torque limiter typically comprises a spring-loaded coupling and friction clutch.

Autopilot engagement

Before the autopilot is engaged, and control of the aeroplane is transferred from manual to automatic, it is important that a number of conditions are satisfied to ensure that the changeover occurs without hazard to flight. For example, the trim of the aircraft must be set by the pilot to avoid any possibility of sudden attitude change when the autopilot is engaged. Similarly, all power supplies to the autopilot system must be operational and a host of

operational parameters must be met. To ensure that it is impossible to engage the autopilot until all requirements are satisfied a system of interlocks is interposed between the autopilot engage switch and its electrical supply. These interlocks take the form of relays and switches that only close when parameters are satisfactory. Since they are connected in series, they must all close before the autopilot can be engaged.

Manual inputs

Mention has been made of the simplest form of manual manoeuvring input to the autopilot, in the form of a rotary knob used by the pilot to bias the inner loops to change the roll or pitch attitude of the aircraft. This type of input control may still be found on the automatic flight system control panel of many aircraft types. In modern transport aircraft, however, it is much more common to apply roll and pitch manoeuvring inputs to the autopilot by means of the control yoke. There are two methods of achieving this, known as control wheel steering (CWS) and touch control steering (TCS).

- **Control wheel steering**. When the autopilot is engaged the pilots can override it, without disengaging, by applying normal manoeuvring force to the control wheel or column. Upon release of the control wheel the autopilot will hold the aircraft at its new attitude and in some cases, if the bank angle is less than 5°, roll the aircraft wings level and hold the new heading until a new automatic flight mode is set on the control panel.
- **Touch control steering**. With this system a thumb switch on the control column is depressed to disengage the autopilot whilst the pilot manoeuvres the aircraft. When the thumb switch is released the autopilot re-engages to hold the aircraft at its new attitude until an automatic flight mode is reselected.

JAR-OPS requirements for autopilots

- Each approved autopilot system must be capable of quick and positive disengagement by the pilots, so that it does not interfere with their control of the aircraft.
- Each system must have a means of indicating to the pilot the alignment of actuating devices in relation to the controls they operate.
- Manually operated control must be so positioned as to be readily accessible to the flight crew.
- Each control wheel or yoke must have a quick release control on the side opposite the throttles.
- Attitude controls must operate in the sense and plane of motion to be

achieved and the direction of motion must be clearly indicated on, or adjacent to, the respective control.

- The autopilot system must not be capable of placing hazardous loads on the aircraft or create hazardous deviations from the flight path under any flight condition appropriate to its use.
- When the autopilot system is integrated with other systems or auxiliary controls there must be interlocks and sequencing to ensure proper operation. There must also be protection against adverse interaction of components.
- Armed and engaged modes of operation must be indicated to the flight crew.

Automatic flight system

A typical automatic flight system for a passenger transport aircraft in the medium to large range is made up of a number of component systems. These include the flight management system, the flight director system and an autothrottle system. The latter will be described in more detail later in this chapter, but briefly its role is to maintain selected EPR and N_1 conditions at specific flight phases as directed by the flight management computer or as set by the pilots on the autopilot flight director system (AFDS) mode control panel. It will be recalled from Chapter 4 that the current automatic flight and autothrottle status is displayed on the EFIS ADI and HSI display screens.

The AFDS mode control panel is usually situated on the cockpit coaming beneath the windscreen and its function is to provide the pilots with control of the autopilot systems, the flight director, autothrottle settings and altitude alert settings. The design of the AFDS and its control panel will clearly vary according to the size and performance of the aircraft in question. The type described in the following paragraphs is typical of certain marks of Boeing 737, but its features and the general operating parameters are similar to those of most passenger transport turbofan aircraft. This particular system uses two completely independent autopilot systems to allow for system redundancy, but many modern aircraft employ three independent autopilots to improve failure protection even further.

Figure 5.13 shows a diagram of the captain's half of the autopilot FDS control panel.

The control panel

The system uses two independent flight control computers that, in automatic flight, supply pitch and roll commands to the inner loops of the autopilot systems. In manual flight control the computers position the command bars on the captain's and first officer's ADI displays. Each pilot has a flight director selector switch. When switched on, the ADI command bars will

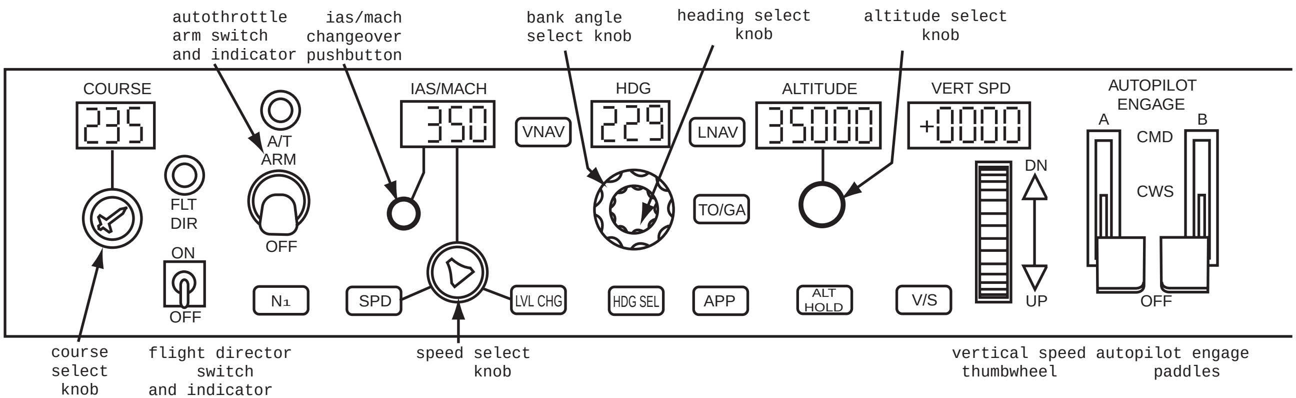

Figure 5.13 AFDS control panel.

appear in certain command modes; when switched off, the command bars will retract out of view. The various mode selector push button switches are depressed for selection and will illuminate to indicate mode selection. Depressing an illuminated switch will deselect that mode. The system will only accept a new mode selection provided that it does not conflict with the mode(s) currently in operation.

Engagement and disengagement of the autopilots is made with paddle switches, one for each autopilot. The paddles have three positions; OFF disengages the respective autopilot, labelled A and B, CWS engages the autopilot but control of flight is by operation of the control wheel and column and CMD is the position for full automatic flight control, enabling all the command modes and CWS operation as required. In all flight phases other than approach (APP) only one autopilot may be engaged at any one time, but approach mode requires both autopilots to be engaged for a fully automatic landing. Command modes may only be armed or selected provided that at least one of the engage paddles is set to CMD and one or both flight directors are switched on. An armed mode is one that has been selected, but will only engage when certain parameters are met.

Autopilot command modes

The following are brief descriptions of each of the command modes of the system under discussion:

- **Vertical navigation (VNAV) mode**. When this selector is depressed the flight management computer commands the AFDS pitch control and autothrottle to follow the selected vertical flight profile programmed into the flight management system. The programmed climb and descent rates, cruise altitudes, speeds and height limitations will be followed through automatic selection of pitch attitude and thrust. With VNAV selected the EFIS ADI will display VNAV PTH or VNAV SPD, depending upon the phase of the planned flight and SPD, N_1, RETARD or ARM for the current autothrottle mode.
- **Lateral navigation (LNAV) mode**. Engagement of LNAV mode causes the flight management computer to command the AFDS roll control to intercept and track the lateral route programmed into the flight management system from waypoint to waypoint. The programme includes all flight procedures such as SIDs, STARS and ILS approach. LNAV will only engage provided that there is a flight path programmed into the flight management system computer. It will automatically disengage if the planned track is not intercepted within certain criteria or if the HDG SEL push button is depressed.
- **N_1 mode**. With N_1 selected the autothrottle system positions the thrust

levers to maintain whatever limiting rpm is set on the flight management computer for the current phase of flight.

- **Speed mode**. With this mode selected the autothrottle system positions the thrust levers to maintain the speed selected with the rotary speed select knob and displayed on the AFDS control panel. The autothrottle system will ensure that the selected speed is achieved without exceeding N_1 limits and will equalise N_1 on both engines provided that it can do so without exceeding 8° difference of thrust lever position.
- **Level change (LVL CHG) mode**. In this mode automatic control of pitch and thrust is co-ordinated for climb or descent to a preselected altitude at preselected airspeed. Before engaging LVL CHG a new altitude is selected with the rotary altitude select knob on the AFDS control panel and this is displayed digitally in the appropriate window on the panel.
- **Heading select (HDG SEL) mode**. A selected heading is made by rotating the heading select knob on the AFDS control panel and is displayed digitally in the HDG window. Depressing the HDG SEL push button will send a roll command to the autopilot to intercept and hold the selected heading. The bank angle during the turn can be controlled with the bank angle select knob, which forms the outer perimeter of the heading select knob.
- **Approach (APP) mode**. With approach mode selected the AFDS is armed to capture and hold the ILS localiser and glideslope. Only when this mode is armed is it possible to engage both autopilots; at any other time moving one autopilot paddle to CMD will automatically disengage the other. To meet the requirements of a fail passive control system (to be explained shortly), both autopilots must be engaged for completion of a fully automatic landing sequence. In this mode the AFDS will command the autopilots through the ILS descent, landing flare, touchdown and roll-out phases. An autoland sequence is described later in this chapter.
- **Take-off/go-around (TO/GA) mode**. The go-around mode is automatically armed when FLARE ARMED is annunciated on the flight mode annunciator and/or EFIS display. Depressing the TO/GA selector push button under these circumstances will engage go-around mode, whereupon the flight director will command a 15° pitch up attitude for a climb on present track to a radio altitude of 400 ft. The autothrottle system will simultaneously command the thrust levers to advance for go-around N_1 rpm. Once 400 ft radio altitude has been passed, other pitch and roll modes may be engaged; prior to that both autopilots must be disengaged if pitch or roll attitude is to be changed.
- **Altitude hold (ALT HOLD) mode**. Selection of altitude hold mode will either maintain the aircraft at the selected altitude or adjust the aircraft's attitude until the selected altitude is attained, in either case by pitch commands. If a new altitude is selected with ALT HOLD engaged, the

select push button will illuminate until the new altitude is reached. Alternatively, the new altitude may be selected first and then ALT HOLD engaged. With ALT HOLD engaged, LVL CHG, V/S and VNAV modes are inhibited.

- **Vertical speed (V/S) mode**. In this mode the flight director provides pitch commands to maintain the selected rate of climb or descent and the autothrottle system adjusts the thrust levers to maintain the selected indicated airspeed. Engagement of V/S mode is annunciated on EFIS and/or the flight mode annunciator and the present vertical speed is displayed on the control panel, prefixed by + or − to indicate rate of climb or descent, respectively. The desired vertical speed is set by rotation of a thumbwheel on the mode control panel.

Automatic landing (autoland)

For an automatic flight control system to be capable of automatic landing it must meet certain criteria. As has already been stated, it must contain a minimum of two independent autopilot systems and, in addition, it must satisfy the following safety requirements:

- The response of the system must be such that there will be no deviation from the flight path in the event of external disturbance such as turbulence or windshear.
- Control system faults must be indicated to the pilot as a warning or alert.
- Control system failures must not cause the aircraft to deviate from the flight path.
- The flight control system must have sufficient control authority to ensure accurate maintenance of the flight path.
- The effect of a servomotor runaway must be limited, such that safe recovery by the pilot is not jeopardised.
- The automatic flight control system must not prevent completion of the intended landing manoeuvre in the event of a system failure.

The above criteria are met by incorporating redundancy in the flight control system through duplication or triplication of the autopilot systems, so that a single failure within the system has a minimal effect on the overall aircraft performance during approach and landing. Depending upon the degree of redundancy, the autoland system is classified as being either a fail passive (fail soft) system or a fail operational (fail active) system.

Fail passive system

An automatic flight system is considered to be fail passive if there is no significant deviation from the flight path, or out-of-trim condition, following

a failure within the system, but the landing cannot be completed under automatic control. In simple terms it means that, if one of the autopilots fails, the other will disengage (since two are required for completion of an automatic landing), but there will be nothing to prevent the pilot completing the landing manually. It follows from this that an automatic flight control system incorporating two independent autopilots must be a fail passive system. Furthermore, a self-monitoring system is essential to ensure that both autopilots are in agreement at all times. These are the minimum requirements for the multiplex type of control system necessary to meet autoland certification.

In the event of failure of either autopilot or the monitoring system during an automatic approach, the approach will continue on one autopilot, but automatic landing is no longer possible. The flight crew must take over manual control and revert to category 1 minima for landing, either continuing the landing or executing go-around procedures at decision height. The single autopilot will disengage automatically at about 350 ft radio altitude.

Fail operational system

In order for a landing to be completed automatically, following a failure within the system, it follows that there must be at least three independent autopilots and two independent monitoring systems. A single failure in either of these will render the system fail passive, but it still has sufficient redundancy to meet the criteria for completion of an automatic landing. In a fail operational system all the autopilots and self-monitoring systems must be engaged for an automatic approach and landing.

The EFIS display indicates the number of engaged autopilots, with a caption reading LAND 3 indicating three autopilots engaged and a fail operational system, LAND 2 a fail passive system with two autopilots engaged and LAND 1 a passive failure with automatic disengagement pending and completion of the automatic landing impossible. At all other flight phases only one autopilot may be engaged at a time.

In the case of fail operational systems there is a specified alert height, determined by the performance of the aircraft and the automatic landing system. Failure of a redundant autopilot or monitoring system above this radio altitude will result in discontinuance of the automatic landing. If failure occurs below alert height the automatic landing is continued on the remaining autopilot, on the basis that manual reversion is more hazardous at this late phase (typically below decision height for manual completion of landing) than to continue in automatic control.

Automatic landing sequence

The sequence of events during an automatic landing is illustrated in Figure 5.14. The radio altitudes for the events during the final stages of the

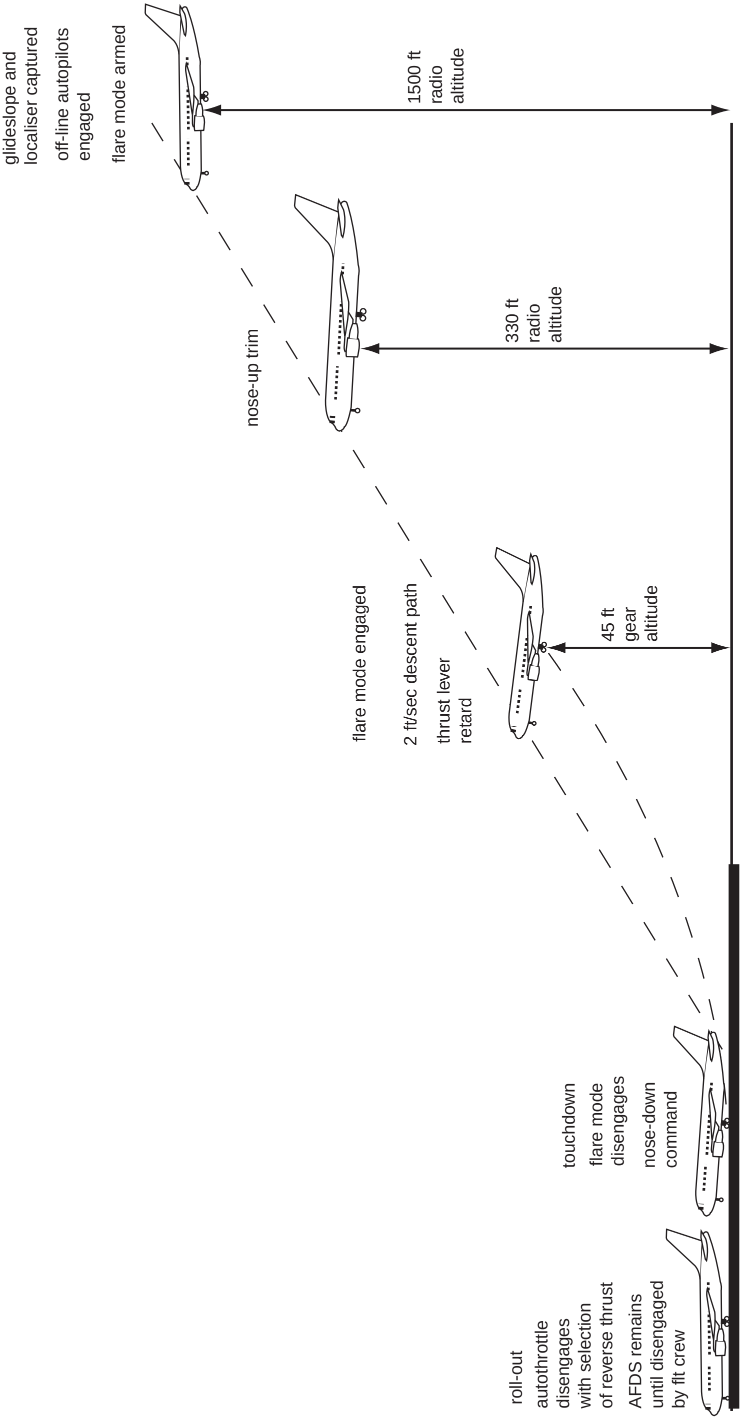

Figure 5.14 Automatic landing sequence.

approach to touchdown will vary according to aircraft size and performance, but the sequence is typical for most aircraft types.

During the descent from the cruise, approach mode is selected by depressing the APP pushbutton and this arms the off-line autopilots; the second in the case of a fail passive system and the second and third in the case of a fail operational system. At the same time the ILS glideslope and localiser channels become the armed pitch and roll modes.

The radio altimeter becomes effective at, typically, 2500 ft agl and provides all height measurements for the automatic flight control system from then until touchdown. At 1500 ft radio altitude, provided that the localiser and glideslope beams have been captured, the off-line autopilots engage and LAND 2 or LAND 3 is displayed on the autoland status annunciation, depending on the number of engaged systems. The aircraft continues to be flown by one autopilot, with the remainder performing a comparative function, overseen by the monitoring system. If these sequences have been satisfactorily completed, FLARE mode becomes armed and the glideslope and localiser beams become the engaged pitch and roll command modes, maintaining the aircraft on the glidepath centre line.

When the aircraft has descended to 330 ft radio altitude, the AFCS commands a nose-up trim adjustment, with pitch control being maintained through the elevators. When the main landing gear is 45 ft above ground level, as measured by the radio altimeter and adjusted to take account of the height difference between the radio altimeter transceiver and the main gear, FLARE mode engages and provides pitch commands. Roll commands are still from the localiser, to keep the aircraft on the centre line of the glidepath. The aircraft now follows a 2 ft per second descent path, rather than the glideslope beam, and the autothrottle system begins retarding the thrust levers to control airspeed for the touchdown.

Just prior to touchdown, at about 5 ft gear altitude, flare mode disengages and touchdown and roll-out modes engage. At approximately 1 ft gear altitude the AFCS commands a decrease in pitch attitude to 2° nose-up and, at touchdown, the elevators are adjusted to lower the nose and bring the nose wheels into contact with the runway. Selection of reverse thrust by the pilot disengages the autothrottle system, but the AFCS remains in control of the roll-out until disengaged by the flight crew.

Automatic thrust control (autothrottle)

The autothrottle system receives its commands from an autothrottle computer, which is linked to the flight management and flight control computers and operates the thrust levers through servo-actuators. Its function is to control the thrust in terms of engine pressure ratio (EPR), HP spool rpm (N_1) or the aircraft's flight speed. Its primary function is to operate in conjunction

with the automatic flight control system in its VNAV and approach modes, to attain a required airspeed and to maintain the programmed vertical flight path. The autothrottle system is armed by operation of a switch on the mode control panel of the automatic flight control system and is controlled through this panel during automatic flight. Figure 5.15 shows in block schematic format the signal interfacing between the autothrottle computer and other systems.

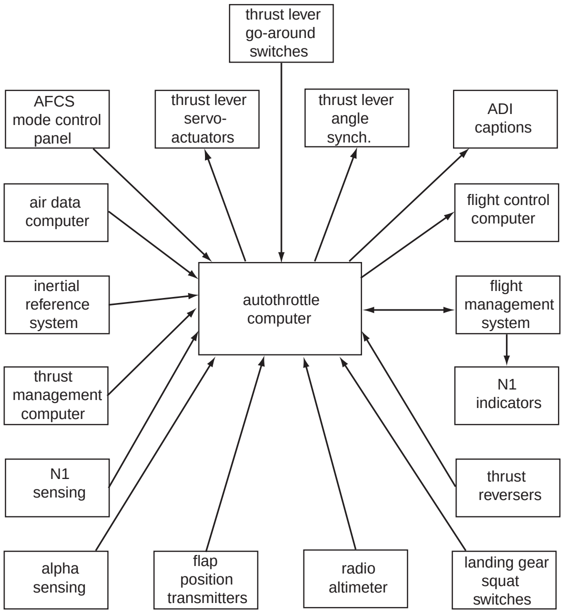

Figure 5.15 Autothrottle signal interfacing.

Operating modes

The autothrottle system operates in one of three possible modes: take-off, speed control and go-around.

Take-off mode

Before commencing the take-off the flight management system is engaged and its computer supplies the N_1 limits for each stage of the flight profile, together with a selected, or 'target', N_1 rpm. These values are displayed as markers on the N_1 indicators of the engine displays. Switching the autothrottle engage switch on the mode control panel to ARM will arm the autothrottle system for take-off and this will be annunciated on the EFIS, or other, display. The thrust lever servo-actuators are engaged by pressing switches mounted on the thrust levers, known as take-off/go-around

switches. An example of these is illustrated in Figure 5.16. Once this has been done, the servo-actuators advance the thrust levers at a preset rate in order to reach the position for take-off N_1 by the time a specific speed has been reached on the take-off roll. For example, the advance rate for the thrust levers might be 15° per second to ensure all engines have reached take-off N_1 before the aircraft has reached a speed of 60 knots. When this target speed has been exceeded by a preset amount, autothrottle movement of the thrust levers is interrupted by a speed detection circuit and the levers are held at their current position, a condition known as throttle hold (THR HOLD). Should the speed detection circuit fail, a back-up system, activated by the main landing gear 'squat' microswitches, will operate to instate throttle hold shortly after the aircraft lifts off. At a radio altitude of 400 ft the autothrottle system arms to control N_1 for the vertical profile of the remainder of the flight and the automatic flight control system takes over control of the autothrottle system.

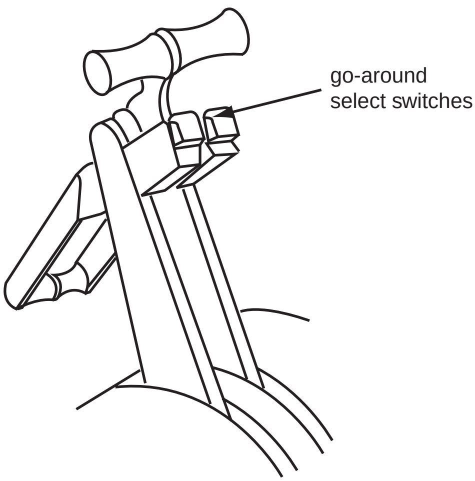

Figure 5.16 Thrust lever go-around switches.

Speed control mode

Speed control mode is selected through the mode control panel of the automatic flight control system, either by the pilot pressing the SPD push button switch or automatically if the system is in other than a speed mode (e.g. VNAV). In either case, the autothrottle system will command the thrust lever actuators to adjust the levers until the IAS or mach No. selected has been reached and held. The autothrottle system controls airspeed/mach to maximum and minimum safe values, regardless of the selected airspeed/mach, and it prevents the angle of attack (alpha angle) from exceeding a safe value. Minimum safe airspeed and maximum safe alpha are computed from

data received from the flap position sensors and angle of attack sensors. Under VNAV control mode the autothrottle system begins to retard the thrust levers at the top of descent at, typically, 2° per second until they either reach the idle stop or are arrested by pilot intervention. During retardation a RETARD annunciation appears, followed by ARM when the retarding movement ceases, to indicate that speed mode is armed. At glideslope capture the AFCS mode changes from VNAV to approach (APP) and the autothrottle engaged mode changes to speed, with the displayed speed being that computed by the flight management system. The gain, or sensitivity, of the autothrottle system is increased for greater precision of speed control during the approach. During the flare manoeuvre the thrust levers are retarded at a rate computed to reach idle in 6 seconds. Immediately after the landing gear microswitches have indicated touchdown, further thrust lever retardation is initiated by the autothrottle system, until it automatically disengages 2 seconds after touchdown.

Go-around mode

Depressing a take-off/go-around switch on one of the thrust levers with the autothrottle system engaged and the aircraft below 2000 ft radio altitude will initiate advance of the levers until they reach the position for reduced go-around thrust. The caption GA will immediately appear on the ADI and the flight management system computer will calculate the full go-around thrust rating determined by the present aircraft all-up weight and the density altitude. A second depression of the thrust lever-mounted TO/GA switch will now advance the thrust levers to increase engine thrust to the full go-around thrust rating. The AFCS will generate the pitch-up and wings level commands necessary to establish the aircraft in the go-around climb-out.

Full flight regime autothrottle system (FFRATS)

This system performs all the functions described above and additionally provides engine overboost protection and selection of variable engine rating. The system monitors demands on the engines made by air conditioning system and anti-icing system air bleeds and adjusts the engine pressure ratio (EPR) limits to suit. Most large modern passenger transport aircraft are fitted with an autothrottle system meeting the FFRATS specifications.

Thrust computation

In order to achieve maximum fuel economy and to prolong engine life, advanced aircraft turbine engines utilise electronic engine control systems. Full-authority electronic engine control systems receive data from the aircraft and engine systems to enable safe and efficient operation of the thrust

management system over the entire operating range of the engines. One aspect of such a system is computation of the optimum and maximum thrust requirement for every condition of flight. The computed total air temperature (TAT) and measured pressure altitude are used to compute the optimum and limiting engine pressure ratio (EPR) for the current flight phase. EPR is the ratio of HP turbine exhaust pressure to LP compressor inlet pressure and has been found to be directly proportional to the thrust delivered by the engine.

The computed EPR for the current flight phase is presented on an indicator on the flight deck, which typically displays TAT, the current flight mode (e.g. take-off, climb, cruise, etc.) and the EPR limit for that mode. The actual EPR, with limit and target markers, continues to be indicated on the engine monitoring display (e.g. EICAS). Examples of EPR indicators are shown in Figure 5.17.

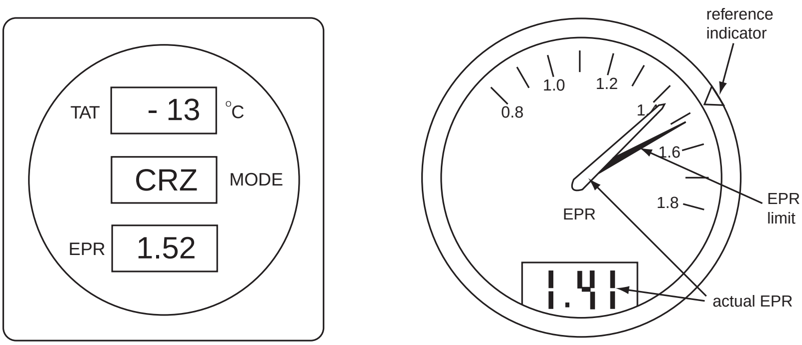

Figure 5.17 EPR indicators.

The flight mode for which EPR computation is required is selected by the pilot through an EPR limit control panel and this is fed to the EPR computer. Typical modes for EPR limit computation are climb (CLB), economy (CON), cruise (CRZ), top-of-descent (TOD) and go-around (GA). When the automatic flight control system is in use, go-around EPR limit will automatically display as the glideslope is captured. Additionally, the panel may contain thrust rating selector switches, with which the pilot can command the computer to calculate the EPR for specific engine performance ratings. The system incorporates a test function for preflight testing. In the event of system failure or electrical power loss, a warning flag obscures the EPR limit indicator.

Sample questions

1. The display screen of a flight management system multi-purpose control and display unit:

   a. Provides an analogue display of flight progress?
   b. Provides an alpha-numeric display of flight plan data?
   c. Is divided into two fields, the left field providing information in alpha-numeric form and the right field providing the same information in analogue form?
   d. Provides 24 lines of information, 14 characters per line?

2. The background tape of a flight director ADI normally has .... freedom of movement in roll and .... freedom of movement in pitch:

   a. 360° ±90°?
   b. ±90° 360°?
   c. 90° 90°?
   d. 360° 360°?

3. The aircraft symbol on a flight director HSI:

   a. Rotates with the compass rose?
   b. Aligns with the selected VOR radial?
   c. Aligns with the selected heading?
   d. Is fixed at the centre of the display?

4. With VOR selected and the HSI lateral deviation bar displaced one dot to the right, the aircraft is approximately .... from the radial:

   a. 5°?
   b. 5 nm?
   c. $1\frac{1}{4}°$?
   d. $2\frac{1}{2}°$?

5. The warning flags on a flight director ADI will indicate failure of (answer a, b, c or d):

   1. glideslope receiver
   2. FD vertical gyro
   3. FD computer
   4. MHRS signal
   5. VOR or localiser signal

   a. 1, 2, 3, 4, 5?
   b. 1, 3, 4, 5?

c. 1, 2, 3?
d. 3, 4, 5?

6. With the flight director in VOR/NAV mode, the ADI command bars provide:

   a. Guidance in pitch only?
   b. Guidance in pitch and roll?
   c. No guidance, since they are retracted in this mode?
   d. Guidance in roll only?

7. A two-axis autopilot has:

   a. A single inner loop and two outer loops?
   b. Two inner loops and two outer loops?
   c. Two inner loops?
   d. One inner loop?

8. The data supplied to an autopilot system from the central air data computer are known as:

   a. Manual data?
   b. Manometric data?
   c. Monometric data?
   d. Aerodynamic data?

9. The function of a torque limiter in the servo-drive of a flying control surface is to:

   a. Prevent excess rate of movement of the surface?
   b. Prevent slip in the drive system?
   c. Control the rate of movement of the surface?
   d. Prevent over-torquing of the servo-motor?

10. Control wheel steering (CWS) is engaged by:

    a. Rotating a knob on the AFDS control panel?
    b. Applying normal manoeuvring force to the pilot's controls?
    c. Operating a thumb switch on the control wheel?
    d. Moving the autopilot engage paddles to OFF?

11. With the automatic flight director system in VNAV and LNAV mode, engaging a second autopilot will:

    a. Improve the sensitivity of the automatic flight system?
    b. Automatically engage the autothrottle system?

c. Disengage the active autopilot?
d. Automatically change the control mode to approach mode?

12. LVL CHG, V/S and VNAV modes are inhibited when .... is engaged:

a. HDG SEL?
b. LNAV?
c. N_1?
d. ALT HOLD?

13. A fail operational automatic landing system:

a. Requires two independent autopilots and one monitoring system?
b. Is one which will not continue with an automatic landing in the event of a single failure within the system?
c. Will abort the landing if a failure occurs below alert height?
d. Requires at least three independent autopilots and two independent monitoring systems?

14. For an automatic landing to continue below 1500 ft, which of the following conditions must be satisfied (answer a, b, c or d):

1. Glideslope capture
2. Localiser capture
3. Radio altimeter serviceable
4. Off-line autopilots engaged
5. FLARE mode armed

a. 1, 2, 3, 4?
b. 1, 2, 4, 5?
c. 1, 2, 3, 4, 5?
d. 1, 2, 4?

15. Autothrottle take-off mode is engaged by:

a. A push button switch on the AFDS mode control panel?
b. Press switches on the thrust levers?
c. A lever switch on the AFDS mode control panel?
d. A push button switch on the control yoke?

16. The factors necessary for computation of EPR limit are:

a. TAT, pressure altitude, flight mode?
b. TAT, pressure altitude?
c. N_1, TAT?
d. N_1, TAT, flight mode?

Chapter 6
In-Flight Protection Systems

Flight envelope protection

Every aircraft design is tested mathematically and in flight to determine the limits of pitch, roll, yaw, angle of attack and 'g' force that the airframe can withstand in flight without suffering structural damage. These limits then form what is known as the flight envelope for that particular design, within which the aircraft can be safely operated. With a conventionally controlled aircraft it is clearly possible to exceed the limits of the flight envelope by applying excessive control movements.

As a means of eliminating the possibility of exceeding the limits of the flight envelope through human error, the fly-by-wire system of flight control has been developed. With such a system, the pilot's control demands are transmitted to computers that are programmed to respond with signals to the appropriate flying control servo-actuators which will limit their rate of movement, thus ensuring that the aircraft response remains within the limits of the flight envelope.

In the Airbus series of aircraft, beginning with the A320, the fly-by-wire concept has been developed to the extent that the fly-by-wire computers have complete control over each of the flying control surfaces, in response to pilot demands from a small side-stick type of control. The response of the computerised system to pilot inputs must be the same as in a conventional direct control system, but the nature of the inputs is more complex because the pilot can demand, for example, a rate of pitch or roll instead of a simple control movement. This type of fly-by-wire system is known as an active control system.

Given that there is no provision for reversion to manual control in these aircraft, it is clearly vital that there must be a degree of redundancy in the fly-by-wire control system sufficient to sustain failure of a computer without degradation of aircraft control. This is achieved by employing a number of computers in an active control system, such that no single computer can command a control surface movement without being monitored by at least one other. The A320 aircraft employs seven computers, connected by a data bus, to control the elevators, ailerons, horizontal stabiliser, spoilers and rudder. Two computers control the elevators, ailerons and the horizontal

stabiliser and are known as the elevator/aileron computers (ELAC). Three computers control the spoilers, elevators and horizontal stabiliser and are known as the spoiler/elevator computers (SEC). It can be seen that control of the aircraft in pitch and roll is shared between the two computer systems so that a fault in one system will not adversely affect the aircraft control. A third pair of computers controls the aircraft in yaw, known as the flight augmentation computers (FAC).

Yaw damper

Swept-wing aircraft, to a greater or lesser extent, exhibit a tendency to develop an oscillatory motion in flight, following a disturbance, which is a combination of yawing and rolling and is known as 'Dutch roll'. In many cases the motion is damped out naturally by the 'weathercocking' effect of the vertical stabiliser and the aircraft quickly returns to steady flight. However, swept-wing aircraft exhibit less natural damping because the yawing motion initiates rolling and, in some cases, the oscillations increase if unchecked, especially at lower flight speeds. The tendency can only be checked by deflection of the rudder and to achieve this manually throughout a long flight would place considerable strain on the pilot.

In aircraft that are susceptible to Dutch roll it is usual to install a yaw damping system that automatically applies rudder deflection to control the yawing tendency. The system comprises the third (yaw) axis of an autopilot system and can be operated in either automatic or manual flight control. A block schematic diagram of a yaw damping system is shown in Figure 6.1.

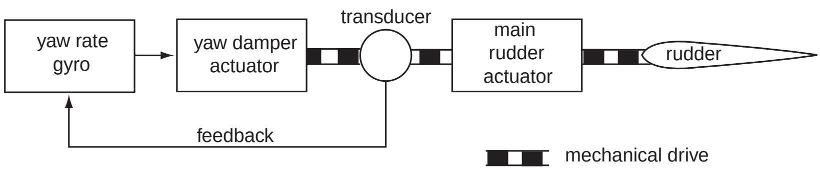

Figure 6.1 Yaw damping system.

Motion about the aircraft's yaw axis is sensed by a rate gyroscope situated in a coupler unit and powered from the aircraft's 115 V a.c. electrical system. Output signals from the yaw rate gyro are amplified and filtered to remove frequencies not associated with Dutch roll, and transmitted to an hydraulic transfer valve in the rudder power control unit (PCU). Movement of this valve directs hydraulic pressure to the yaw damper actuator. The resultant movement of a piston in the yaw damper actuator operates a control valve in the main rudder actuator, which moves the rudder in the required direction to correct the yawing tendency sensed by the rate gyro. The yaw damper

piston motion is sensed by a transducer, known as a linear voltage displacement transmitter (LVDT), and fed back to the gyro unit. When the actuator piston has moved by the amount demanded, this feedback of rudder position cancels the gyro output and rudder movement is arrested. When the Dutch roll oscillations have ceased, the LVDT signal is integrated in the rate gyro coupler unit, to produce an output signal returning the rudder to its neutral, centralised position.

Yaw damper indicator

On many aircraft equipped with a yaw damping system the operation of the yaw damper is indicated on the EADI in conjunction with the rate of turn indicator. This receives a signal from the yaw rate gyro. Whenever the gyro precesses, the signal causes the rate of turn indicator to move away from its neutral position. Rudder movement is displayed on a control position indicator.

Pilot operation of the rudder is by direct linkage to the main rudder actuator and is therefore independent of the yaw damping system. Rudder movements commanded by the yaw damping system are not transmitted back to the rudder pedals.

Automatic pitch trim

In an aircraft equipped with a movable horizontal stabiliser (trimmable stabiliser) and elevator for pitch control, pitch trim is normally adjusted by first moving the elevators, followed by trimming the horizontal stabiliser until the elevator is returned to the neutral, centralised position. The normal action of an autopilot system in compensating for an out of trim condition in pitch is to move the elevators until the condition is corrected.

The disadvantage of this system is that, once the elevators are deflected, the amount of remaining movement in that direction is limited and control authority in pitch is reduced. Furthermore, with the elevators deflected from their centralised position, drag is increased, with the obvious adverse effects upon fuel economy and, ultimately, range and endurance.

Consequently, it is not uncommon for aircraft with the stabiliser/elevator configuration to incorporate a system additional to the automatic flight system, which will automatically adjust the horizontal stabiliser until the elevators are restored to the neutral position. Such a system is known as an automatic stabiliser trim (AUTO STAB TRIM) system and it is usually engaged automatically with autopilot engagement. It is a requirement of autopilot engagement that the automatic stabiliser trim system must be operational.

The degree of elevator deflection necessary will depend on airspeed and

the automatic stabiliser trim controls incorporate a feel unit which adjusts the trimming signal according to sensed dynamic pressure.

Figure 6.2 illustrates schematically an automatic stabiliser trim system. Pitch commands from the autopilot or from manual inputs are sent to the powered control unit, which deflects the elevators through a screwjack, either up or down depending upon the pitch attitude change required. The automatic trim system will then move the horizontal stabiliser, through the trim actuator and screwjack to apply the nose-up or nose-down trim adjustment initially required. As the stabiliser takes up its new position, its motion is mechanically transmitted to a feel and centring unit and a neutral shift sensor. The deflection of the stabiliser removes the need for elevator deflection and the neutral shift sensor sends a feedback signal to the elevator PCU, removing elevator deflection as stabiliser deflection increases, until the elevator and stabiliser are centralised.

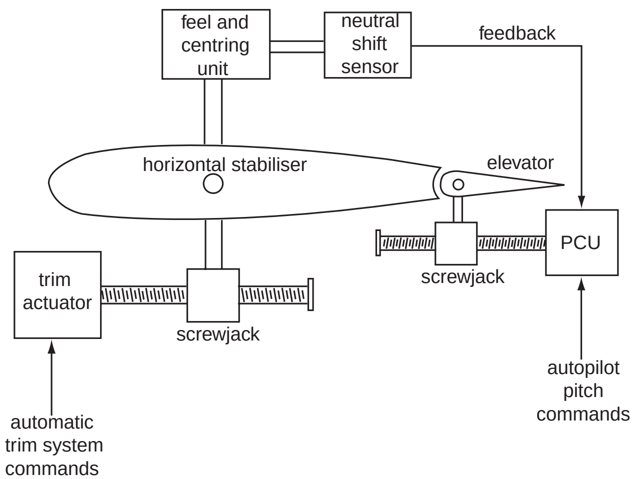

Figure 6.2 Automatic stabiliser trim unit.

In aircraft with a fixed horizontal stabiliser, the pitch trim is achieved by means of elevator trim tabs, which are deflected to assist the elevators, to relieve the aerodynamic loads and some of the drag created by elevator deflection. Automatic pitch trim control is accomplished by means of a separate elevator trim tab servo-actuator coupled to the trim tabs and working in parallel with the elevator servo-actuator. Figure 6.3 illustrates schematically a system sometimes used in conjunction with small aircraft automatic flight control systems.

A sliding bar on a mounting attached to the airframe is connected to a capstan, positioned between the elevator 'up' and 'down' control cables. The cables are lightly tensioned by pulleys so that, when the elevator is in the

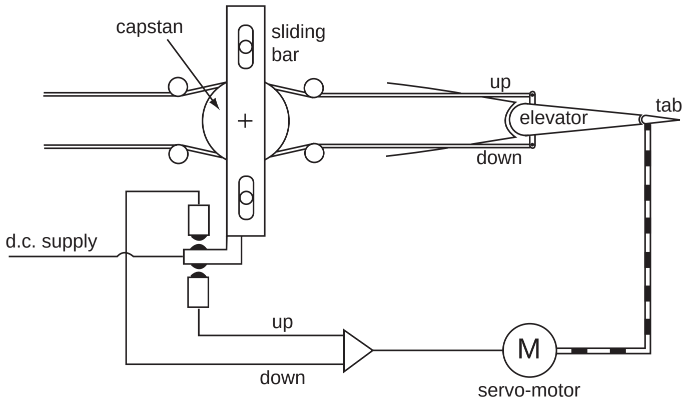

Figure 6.3 Automatic trim tab pitch trim system.

neutral (streamlined) position, the capstan and bar are centralised between the cables. An electrical contact attached to the sliding bar is supplied from the aircraft's d.c. bus bar. Adjustable contacts fixed to the mounting are connected to the 'up' and 'down' field windings of a reversible d.c. motor, which is the trim tab servo-motor.

Let us suppose that the autopilot has demanded a nose-down pitch. The elevator actuator will deflect the elevator down, through the control cables, tensioning the down cable and relieving tension on the up cable. The difference in cable tension will force the sliding bar upward and electrical contact will be made with the trim tab 'down' line, supplying the 'down' field coil of the trim tab servo-motor and driving the tab down to reduce the aerodynamic force on the elevator. As the load on the elevator decreases, the tension of the elevator cables will once again equalise and the sliding bar will return to the centralised position, cutting off supply to the trim tab servo-motor.

Automatic pitch trim systems normally include warnings and alerts in the event of system failure. These typically take the form of warning lights or captions and may include an aural alert should a runaway condition, resulting in excessive trim input, occur. In the case of the automatic stabiliser trim system, there is always a trim indicator on the flight deck.

Warnings general

Flight warning system

The function of a flight warning system is to alert the pilots to the existence of an abnormal situation that requires action; it also identifies the nature and location of the failure or condition. Warnings may be aural or visual, or a

combination of both. Aural warnings may be in the form of a klaxon alarm, a bell or chime or, in some cases, a verbal message. Visual warnings and cautions may be illuminated lights or captions, red for those requiring immediate attention and amber for those requiring urgent attention, and they may also appear as printed messages on the EFIS, EICAS or ECAM displays. Certain hazardous situations, such as a fire warning, are indicated by a general alert in the form of a red master warning light accompanied by an aural alarm. The location of the incident will be indicated separately, for example on an annunciator panel or the relevant electronic display. Annunciator panels usually incorporate other attention-getting lights in blue, green and white to notify the pilots of system availability and status, in much the same manner as the EICAS and ECAM display systems.

In many large aircraft types the flight warning system is computerised and an example of such a system is illustrated in block schematic form in Figure 6.4.

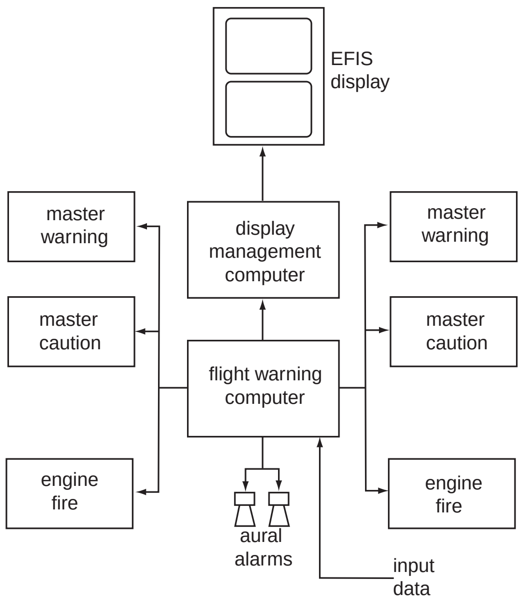

Figure 6.4 Flight warning system.

Altitude alert system

The function of the altitude alert system, as distinct from the ground proximity warning system (GPWS), is to alert the pilots both aurally and visually when the aircraft departs from or approaches a selected altitude on the automatic flight director system. When the aircraft is approaching a selected altitude, and it is within 900 ft of that altitude, the amber altitude

alert lights illuminate and an aural chime alert sounds for a 2-second period. The alert lights remain steadily illuminated until the aircraft is within 300 ft of the selected altitude.

Should the aircraft deviate from a selected altitude by 300 ft the amber altitude alert lights will illuminate and flash repeatedly and the aural chime alert will sound for 2 seconds. The alert lights will continue to flash until the aircraft has deviated from the selected altitude by 900 ft, or has returned to it. Deviation from the selected altitude will not be alerted when the landing gear is extended, since this could lead to confusion with the GPWS alerts.

An altitude alert system is a JAR requirement for all turbojet passenger aircraft and for turboprop aircraft over 5700 kg take-off weight and capable of seating more than nine passengers.

Radio altimeter

The function of the radio altimeter is to measure and display the vertical distance between the aircraft and the ground directly beneath it. It is important to remember that, whilst it is very accurate, it only measures vertical distance and is incapable of measuring terrain clearance ahead of the aircraft.

Principle of operation

Radio altimeters for civil use operate in the SHF band within the frequency range of 4200 MHz to 4400 MHz. A second frequency range of 1600 MHz to 1700 MHz, in the UHF band, is also reserved for radio altimeter operation, but is not used by civil aircraft. The principle of operation is to continuously transmit a variable frequency signal in a relatively narrow beam vertically downward. The signal is reflected from the ground and received at the radio altimeter receiver, located separately from the transmitter. Since it takes a finite time for the signal to travel to the ground and return, and given that the transmitted signal frequency is continuously changing, it follows that the received frequency will differ from the transmitted frequency. The difference between the received and transmitted frequencies will vary as the aircraft height varies, and the time taken for the signal to travel to the ground and back varies. It is the frequency difference that is used to determine the aircraft height above the ground at any instant, using the speed of propagation of the radio beam and the rate of change of transmitted frequency, which are both known.

The transmitted signal is modulated to sweep over a frequency range of, typically, 100 MHz around 500 times per second. This is a deliberately low sweep rate, designed to avoid height ambiguity which might occur at a

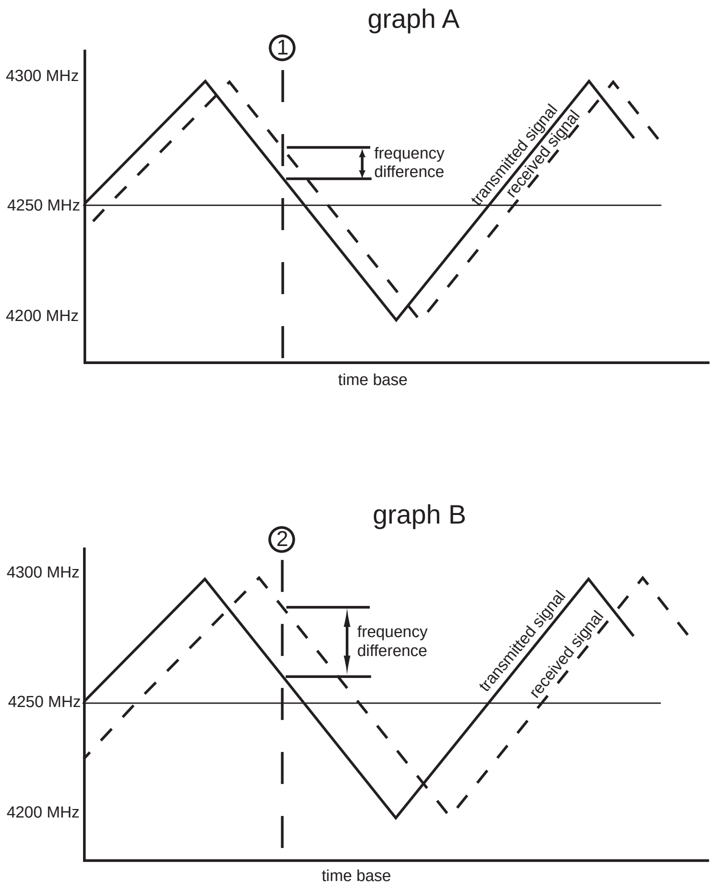

Figure 6.5 Radio altimeter principle of operation.

higher rate of frequency change. The concept of the radio altimeter principle of operation is illustrated graphically in Figure 6.5.

In the example in Figure 6.5 the frequency is being modulated to sweep over the range 4200 MHz to 4300 MHz at a constant rate. The solid line represents the transmitted frequency and the dotted line the received frequency. At a particular point in time, represented by the vertical line annotated 1 in graph A, it will be seen that there is a difference between the two frequencies; this difference is directly proportional to the aircraft height and is represented as such on the radio altimeter display. In graph B, the vertical line annotated 2 represents an instant in time with the aircraft at a greater height and consequently the difference between transmitted and received frequencies is greater. The frequency difference is exaggerated in these diagrams; in reality it is quite small, even at the usual maximum radio altimeter operating height of 2500 ft.

System components

The principle components of a radio altimeter system are the transmitter, the receiver and the display unit. However, as we have seen in previous chapters, the display is incorporated in the ADI displays of EFIS or flight director equipped aircraft. An example of the type of display unit found in aircraft not so equipped is shown in Figure 6.6. It will be seen that the instrument includes a decision height feature that allows the pilot to set the decision height index bug on the face of the instrument. When the pointer reaches the set height during a descent a visual and/or aural warning is activated. The visual warning is usually in the form of a light and the aural warning may be a chime alert or a recorded voice message. In the event of failure of the system due to loss of power, a system or reception fault, a prominent warning flag appears on the face of the instrument. Additionally, the pointer will be obscured on these occasions or when flying above 2500 ft. The pointer will take up a known position when the press-to-test button is depressed. In some displays the instrument scale is logarithmic for heights above 500 ft.

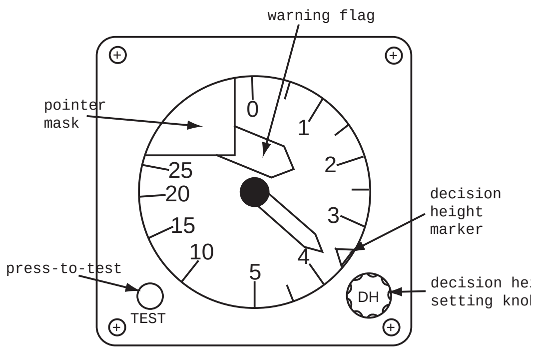

Figure 6.6 Radio altimeter display.

Accuracy

The accuracy of the radio altimeter is given as ± 1 ft or $\pm 3\%$ of the indicated height, whichever is the greater. It can be subject to errors due to reflections from parts of the aircraft structure, such as the landing gear, or to leakage of signals between the transmitting and receiving aerials. The positioning of the aerials is therefore very important and every effort is made by the manufacturer to avoid these errors. It is also conceivable that the receiver might pick up a signal that has been reflected from ground to aircraft more than once, known as a multi-path signal. To a large extent this potentially ambiguous situation is avoided by gain control in the receiver.

The principle use of radio altimeter information in large modern aircraft is in conjunction with the automatic landing system and the ground proximity warning system.

Ground proximity warning system (GPWS)

The purpose of a ground proximity warning system is to provide aural and visual signals to the pilot when the aircraft is in danger of impacting with the ground, unless corrective action is taken. Joint aviation requirements are that all turbine-powered aircraft having a maximum certificated take-off weight greater than 5700 kg and seating for more than nine passengers must be equipped with a GPWS.

There are three types of GPWS currently in use: basic, advanced and enhanced.

Basic GPWS

The basic GPWS has five modes of operation, which require the following source inputs:

- **Radio altimeter**. Accurate measurement of height above ground level is provided by the radio altimeter.
- **Central air data computer**. Barometric pressure is integrated by the GPWS to compute descent rates.
- **ILS glidepath receiver**. The GPWS is required to give warning of descent below the glidepath during a landing approach.
- **Approach configuration**. The landing gear and flaps positions are necessary inputs to the system during the approach to land.

The GPWS must be active between 2450 ft and 50 ft above ground level.

Operating modes

Mode one, excessive descent rate

This mode is activated at and below 2450 ft radio altitude and is designed to alert the pilot to the fact that the aircraft is descending at a rate that is hazardous. The system measures the barometric rate of descent and gives an aural warning in the form of a klaxon type 'whoop-whoop' alarm followed by a verbal 'pull up' message. These are accompanied by a visual warning in the form of a red light with the caption PULL UP prominently displayed within the pilots' normal field of view. At the upper limit of operation of 2450 ft agl, the warning will be given if the descent rate exceeds 7350 ft per

minute. The lower the aircraft is, the more urgent the required response and so the descent rate that will initiate a warning decreases with height. At 1000 ft agl, the warning is triggered by a descent rate of 3000 ft per minute. At the minimum GPWS operating height of 50 ft agl, the trigger value is 1500 ft per minute.

Mode two, excessive terrain closure rate

Mode two is designed to provide warning when the aircraft is in level flight at or below 1800 ft radio altitude and the terrain beneath it is rising. In this mode the input is from the radio altimeter, but the system also takes account of flap position. Mode 2A is active above 790 ft agl and will only provide a warning if the flaps are NOT in the landing configuration and the terrain closure rate is in excess of 2063 ft per minute. Mode 2B becomes active below 790 ft agl and will give a warning regardless of flap position if the terrain closure rate is 3000 ft per minute or greater. As with all five modes, the lower limit of operation is 50 ft agl. The warnings given, in the event of the terrain closure parameters being exceeded, are the same as in mode one.

Mode three, altitude loss after take-off or go-around

This mode is active between 50 ft and 700 ft agl as measured by the radio altimeter. Its purpose is to alert the pilot to accumulated height loss immediately following take-off or during a go-around manoeuvre. The aural and visual warnings will be initiated if a cumulative barometric height loss occurs, the triggering value of which depends upon radio altitude. For example, at 50 ft agl the warnings will be triggered if the cumulative barometric height loss exceeds 10 ft. At the upper level of 700 ft the warnings will not be activated unless the cumulative barometric height loss is in excess of 70 ft. The trigger value increases linearly between the lower and upper limits of operation.

Mode four, unsafe terrain clearance

The purpose of mode four is to warn the pilot if the terrain clearance is inadequate with the aircraft NOT in the landing configuration. It is divided into two parts, mode 4A and mode 4B. Mode 4A is active between 500 ft and 200 ft radio altitude if the landing gear is NOT down and locked. It is triggered by a barometric descent rate of 1900 ft per minute or more. Mode 4B is active between 200 ft and 50 ft radio altitude and will be triggered if the flaps are not in the landing configuration with the landing gear down and locked. The warnings given are the same as for the previous modes.

Mode five, aircraft below the ILS glideslope

Unlike the previous modes, mode five gives an alert as opposed to a warning. The definition of a warning is a command requiring immediate

response in the form of a maximum gradient climb to a safe altitude. An alert is defined as a caution requiring immediate action to correct the flight path or aircraft configuration. Mode five is active between 500 ft and 50 ft radio altitude and will generate an alert if the aircraft is significantly below the ILS glideslope with the landing gear extended. In broad terms, a significant deviation is given as greater than 150 micro amps displacement from the central null at the glideslope receiver. Mode five is inhibited when mode three is active, to avoid confliction between the two. The alert given is visual and takes the form of an amber light, with a ‘below glideslope’ caption, situated adjacent to the red ‘pull up’ warning light. The mode five alert can be inhibited by pressing the amber light.

An example of a GPWS control and display unit is shown in Figure 6.7. System integrity is tested by pressing the red ‘pull up’ light cover, whereupon the ‘pull up’, ‘below glideslope’ and ‘inop’ lights should illuminate. The override switch and light provide the facility to inform the GPWS when the approach flap configuration is to be other than normal, as in an approach with asymmetric power, for example.

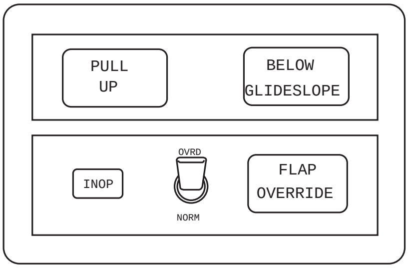

Figure 6.7 GPWS control and display unit.

Advanced GPWS

A disadvantage of the basic GPWS is that it does not differentiate between modes 1 to 4, because it gives the same warning in each case. It is, of course, useful for the pilot to know the cause of the warning when responding to it. Advanced GPWS overcomes this to a large extent by providing an identifying aural alert message to accompany the warning. The amber alert light, which only illuminates in mode 5 of basic GPWS, will illuminate in all cases and consequently carries the caption ‘ground proximity’ instead of ‘below glideslope’.

The alert messages in modes one to three follow the aural ‘whoop-whoop’, ‘pull up’ warning and are as follows:

- **Mode one**. 'Sink rate' repeated.
- **Mode two**. 'Terrain' repeated.
- **Mode three**. 'Don't sink' repeated.
- **Mode four**. The warning message is 'too low terrain', instead of 'pull up'. The alert message in mode 4A is 'too low gear' repeated and in mode 4b it is 'too low flaps' repeated.
- **Mode five**. As with basic GPWS, there is no warning message and the alert message is the same, 'glideslope'.
- **Mode six**. Advanced GPWS has a sixth mode, which is alert only. It differs from the previous five modes in that it alerts the pilot to the fact that the aircraft has reached a certain point in the landing approach, such as decision height, and that the approach may be continued or a missed approach procedure executed, depending upon visibility criteria. The alert message with this mode is 'minimums' repeated.

Advanced GPWS has built-in test equipment (BITE), which continuously monitors the system for faults and these are displayed automatically during flight. The self-testing system display cannot be selected during flight, but is normally part of the pre-flight checks. The ground test will activate all the visual and aural warnings. In the event of unserviceability of the GPWS, the aircraft may only be permitted to fly if the equipment cannot be repaired at the location where the fault is first discovered. In such a case, the aircraft may be flown to an airfield where the fault can be rectified, provided that the journey requires no more than six segments.

Terrain avoidance warning systems (TAWS)

Both basic and advanced GPWS have shortcomings in that they take no account of terrain ahead of the aircraft, the time available to respond to warnings is very limited and the recovery advice is minimal. It would clearly be of great value to the pilot to receive information that a potentially hazardous situation could exist, as far in advance as possible.

With the amount of information stored in flight management computer systems about ground locations of radio navaids, airports, etc., it is a relatively simple task to add a terrain database that covers all the world's major air routes. This database is held in the GPWS computer memory and, combined with the route and location data held in the flight management system computer, gives the system the capability to warn of terrain hazards in the projected flight path. The system currently in use, that complies with TAWS requirements, is known as enhanced GPWS (EGPWS) manufactured by Honeywell.

The inputs to the enhanced ground proximity warning system (EGPWS) are:

- **Radio altimeter**. As in the basic and advanced systems, the radio altimeter determines the vertical height of the aircraft above ground level.
- **Central air data computer (CADC)**. The aircraft's vertical speed is computed from rate of change of barometric pressure and its airspeed from pitot and static pressure. In addition, the CADC provides the EGPWS with static air temperature.
- **VHF receiver**. ILS glideslope information is fed from the VHF navigation radio receiver.
- **Navigation computer**. The aircraft's lateral position and track are supplied continuously from the navigation computer, using radio navigational fixes and IRS computed position information.
- **Flight management system (FMS)**. The FMS provides heading, track, attitude and groundspeed data as generated by the IRS. Where these systems are not available, the information can be provided by a series of accelerometers.

The enhanced system displays surrounding ground features and is able to give significantly greater warning of impact hazard than its predecessors. Terrain details can be presented on the EFIS HSI display or the weather radar display, with colour coding indicating the terrain height relative to the aircraft. Green indicates that the terrain is below the aircraft's projected flight path, terrain that extends above the projected flight path is indicated by yellow and red indicates terrain that is well above the projected flight path. Resolution increases with terrain height. Terrain extending above the projected flight path and which is sufficiently close to generate an alert is painted in solid yellow or red.

Warnings and alerts

When the projected flight path of the aircraft is obstructed by terrain at ranges calculated to present a hazard, caution lights illuminate on the EGPWS panel and a recorded spoken message gives an appropriate aural alert such as 'caution terrain'. If no action is taken to amend the flight path the caution lights are replaced by a red warning light and the aural message becomes 'terrain, terrain' – 'pull up, pull up'.

The general terrain database of the system contains a model of the earth's surface with contours stored in rough detail. Around well used routes and airways the model is more precise and includes obstructions as well as terrain features. In the vicinity of airports it is assumed that aircraft are likely to be descending and so the terrain proximity warning distances are less than those in areas where aircraft would be expected to be operating at altitude. For the areas surrounding airports the database contains precise and up-to-date information of all terrain and obstacles under the approach paths.

EGPWS mode seven

EGPWS has a seventh mode, the function of which is to provide alerts in the event of encountering windshear below a radio altitude of 1500 ft. The system computer compares airspeed and groundspeed to detect significant changes in windspeed and direction. In the event of an increasing headwind, decreasing tailwind and/or severe updraught a caution light will illuminate and the aural message 'caution windshear' will be spoken. In the event of a decreasing headwind, increasing tailwind and/or a severe downdraught, a windshear warning will be given in the form of an illuminated red warning light and the spoken aural message 'windshear, windshear, windshear'.

Traffic Collision Avoidance System (TCAS II)

For many years the separation of aircraft in flight was dependent upon the air traffic control services, using secondary surveillance radar (SSR), and the alertness of the flight crews, using the mark 1 eyeball. As air traffic congestion increased, especially around large airports, and the speed of aircraft made visual warning and avoidance more and more unlikely, the airline operators became highly concerned at the number of near misses and the increasing likelihood of a catastrophic mid-air collision. Fortunately, these circumstances coincided with the rapid advance in computer technology and miniaturisation. It became possible, in theory at least, to extend the SSR principle of a ground station interrogating an airborne transponder and fit aircraft with interrogation equipment as well. Thus, an aircraft in flight would be able to continuously transmit interrogation signals that, when received by another aircraft, would trigger its transponder to respond with details of its altitude. If these, when computed, indicated a potential collision course between the two aircraft, the airborne equipment would supply flight directions to divert the interrogator and avoid conflict.

The operating requirements for a collision avoidance system using these principles were stipulated by ICAO under the title Airborne Collision Avoidance System (ACAS). To date only one system has been introduced into general use that meets these requirements and it is known as the Traffic Alert and Collision Avoidance System (TCAS), developed in the USA. There are two versions of the system, TCAS I and TCAS II, of which only the latter is approved for passenger transport aircraft in the USA and Europe. In Europe the Joint Aviation Authority requires all fixed wing turbine-powered aircraft with seats for more than 30 passengers to be equipped with TCAS II and, by the beginning of 2005, this will be extended to include all aircraft over 5700 kg take-off weight with seats for more than 19 passengers. In the USA, the Federal Aviation Administration (FAA) requires all aircraft with seating for more than 30 passengers to be equipped with serviceable TCAS II.

TCAS I, whilst not approved for use in fixed wing passenger transport aircraft, has proved popular in aircraft that operate at generally lower speeds and altitudes, especially helicopters. It provides the pilot with a visual display of the range and relative bearing of an aircraft on a potential collision course, with 40 seconds in which to acquire the aircraft visually and take avoiding action. Mode C equipped transponders provide altitude information concerning the conflicting traffic and render the system capable of eliminating traffic that has adequate vertical separation from the TCAS display.

TCAS II principle of operation

The TCAS installation consists of a transponder that continuously transmits a pulsed interrogating signal on the standard SSR frequency of 1030 MHz and responds with a coded pulsed signal on the standard SSR frequency of 1090 MHz. The transmitter power required is relatively low, since the range required is limited by the distance necessary to provide adequate warning of potential confliction. The interrogating signal is sent in SSR mode S, which includes mode A interrogation and mode C altitude information (SSR is explained in detail in the *Radio Aids* volume of this series of books). A TCAS II equipped aircraft's transponder receiving this signal will respond with a coded pulsed signal at 1090 MHz containing details of the aircraft altitude. The two receiving antennae are directional and are located one on the top of the aircraft and one underneath. They enable the interrogating aircraft's TCAS computer to calculate relative bearing, and the response time enables it to compute range. From these data the computer can immediately determine whether there is a potential confliction. By integrating the received altitude with time, the interrogating aircraft's computer can calculate and display the rate of change of altitude of the intruder aircraft. All information from the TCAS computer concerning responding aircraft transponders is displayed on an electronic vertical speed indicator (VSI) or on the EFIS display.

The TCAS II system provides a visual display of all responding air traffic. Those responses that are not computed to currently present a threat of collision are known as traffic advisory (TA) messages. When the computer calculates that a confliction is possible it generates resolution advisory (RA) messages describing the necessary avoidance manoeuvre. The resolution advisory only prescribes manoeuvres in the vertical plane, i.e. climb or descend, since the accuracy of the directional antennae is insufficient for safe lateral deviation directions. The resolution advisory messages are passed to the pilot both aurally and visually. The aural messages are in the form of recorded voice messages that indicate the urgency of the pilot response necessary. For example, where an intruder aircraft presenting no current threat exists the traffic advisory (TA) message will be 'traffic, traffic',

drawing the pilot's attention to the visual display and alerting him to the possibility of a resolution advisory (RA). Resolution advisories such as 'climb, climb' or 'descend, descend' require the initiation of a 1500 ft per minute climb or descent within 5 seconds. 'Increase climb/descent' requires the rate of change of vertical speed to be increased to 2500 ft per minute within 2 to 3 seconds and 'climb, climb NOW' or 'descend, descend NOW' requires an immediate reversal of the vertical flight direction.

Visual display

The electronic VSI display referred to previously comprises a 'conventional' VSI scale, albeit electronically generated, around its perimeter and calibrated to show rates of climb in the upper half and rates of descent in the lower. The lateral situation is indicated by an aircraft symbol and azimuth scale in the lower centre of the display and symbols representing intruder aircraft about the upper centre in the respective lateral positions of the transponding intruder aircraft.

There are four alternative intruder symbols, indicating the threat potential presented and the vertical movement of the intruder. Provided that the intruder aircraft is responding with SSR mode C, the symbol will show the relative altitude numerically in hundreds of feet. If the relative altitude of the intruder is changing by more the 500 ft per minute the numeric annotation will be preceded by a plus or minus sign, indicating that it is climbing or descending. This is further emphasized by an accompanying arrow pointing up or down, as appropriate. The shape and colour of the intruder symbol indicates the nature of the advisory message, as shown in the summary below and in Figure 6.8.

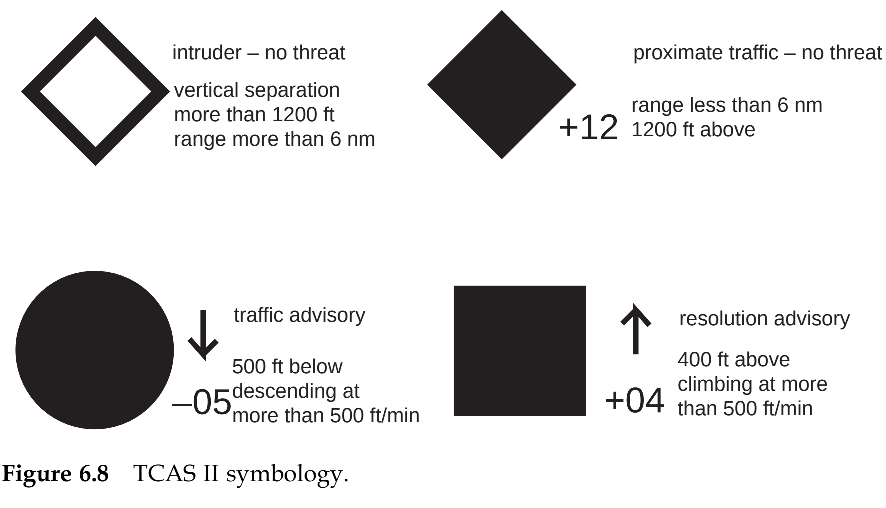

Figure 6.8 TCAS II symbology.

- **No threat**. Traffic beyond 6 nm range or more than 1200 ft vertical separation. An open diamond symbol in white or cyan, with no altitude annotation.
- **Proximate traffic**. Traffic within 6 nm range, but not computed to present a threat. A solid diamond symbol in cyan, with relative altitude annotation.
- **Traffic advisory**. A solid circle in amber, with relative altitude annotation.
- **Resolution advisory**. A solid square in red, with relative altitude annotation.

The circular scale has coloured arcs superimposed upon it to indicate safe and unsafe climb and descent rate areas when a resolution advisory exists. Unsafe areas are indicated by a narrow red arc; the advised rate of change of altitude by a broad green arc. The range of the intruder is indicated digitally outside the circular scale and in analogue form by the position of its symbol relative to the range ring. A typical TCAS II electronic VSI display is shown in Figure 6.9. In this, the equipment has issued a resolution advisory for an intruder at 10 nm range, which is 600 ft above the interrogating aircraft and descending at greater than 500 ft per minute. The broad green arc between 500 and 1000 ft per minute descent rate indicates the RA recommended safe descent rate and the narrow (red) arc the unsafe area.

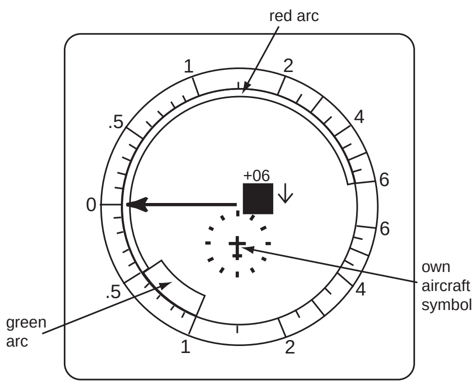

Figure 6.9 TCAS II VSI display.

In EFIS equipped aircraft the intruder symbols are displayed on the horizontal situation indicator in the PLAN and expanded modes and the RA avoidance manoeuvre is shown by the command bars on the ADI display.

When both the interrogating and the intruder aircraft are equipped with TCAS II and SSR mode S capability, the two TCAS computers are able to coordinate the resolution advisories in each aircraft to achieve optimum

separation, with the least disruption to either. This is designed to ensure that both flight crews do not take the same avoiding action and worsen the danger of collision.

Crew response

The system is so designed that, provided pilot response to an RA is taken within the time limits of 5 seconds and 2 to 3 seconds referred to above, the altitude change to avoid confliction should not exceed 500 ft. If the aircraft is under Air Traffic control when a TCAS resolution advisory is received, the pilot is required to obey the TCAS command and inform ATC 'TCAS climb/descent' as appropriate. Upon receipt of the TCAS message 'clear of conflict', the aircraft must be returned to the ATC assigned flight level. Should a manoeuvre instruction be received from both TCAS and ATC simultaneously, the pilot is required to obey the TCAS instruction and advise ATC accordingly.

At radio altitudes of less than 1000 ft the TCAS will not give a resolution advisory involving a descent, and below 1800 ft agl it will not recommend an increased rate of descent, since in either case the hazard to the aircraft in terms of ground impact would be greater than the collision hazard. All resolution advisories are inhibited at radio altitudes of less than 500 ft, and traffic advisories at less than 400 ft.

In order to avoid confusion, TCAS warnings are co-ordinated and prioritised with those of other in-flight protection systems. It is usual for GPWS alerts and warnings to take precedence over TCAS, and a windshear warning will be awarded the highest priority.

Collision warning systems are also described in the *Radio Aids* volume in this *Ground Studies for Pilots* series of books.

Overspeed warning

The function of the overspeed warning system is to provide aural warning, to supplement the visual warnings on the airspeed and mach indicators, when the maximum operating speed V_{mo}/M_{mo} is exceeded. The aural warning given is usually a 'clacker' type of alarm, which can only be stopped by reducing airspeed below the maximum operating limit.

Input data of airspeed and mach number for the overspeed system is obtained from the central air data system. Since the maximum operating airspeed is affected by criteria such as aircraft weight, flap and slat positions and centre of gravity position, these are provided from the flight management system computer.

Visual displays of V_{mo}/M_{mo} are presented on the airspeed indicator, typically in the form of a separate pointer, known as the maximum allowable

pointer, actuated by a static pressure diaphragm and a specially calibrated mechanism. The pointer is usually distinctively marked and is often referred to as the 'barber's pole'. V_{mo} increases with altitude until M_{mo} is reached, so the calibration of the maximum allowable pointer indicates the maximum operating speed adjusted for altitude. M_{mo} is a set value (e.g. 0.84 M) and a mach number operated switch triggers the clacker alarm if this value is reached before V_{mo}, as would typically be the case at altitude. Failure of the V_{mo}/M_{mo} system is indicated by a warning flag on the face of the instrument.

The overspeed warning system incorporates a test function to prove serviceability during preflight checks, which sounds the aural warning.

Stall warning system

The function of a stall warning system is to alert the pilot to the fact that the aircraft is approaching the stalling angle of attack in sufficient time for corrective action to be taken. The actual nature of the warning takes several different forms, depending upon the aircraft type and its behaviour at the point of the stall and thereafter. Since the stall occurs at a known angle of attack, the stall warning sensing device measures aircraft angle of attack and the stall warning system activates an alert before the stalling angle is reached.

Many aircraft provide some warning of the approach of a stall in the form of buffeting, followed by a pitch-down change of attitude as the centre of pressure moves rapidly aft on the wing. The buffeting is felt on the controls in aircraft with controls linked directly to the control surfaces, but with powered flying controls this is less likely to be the case and the natural stall warning may be so slight that the pilot might miss it in conditions of high work-load. In aircraft with the horizontal stabiliser mounted at or near the top of the fin (so-called T-tail aircraft) a condition known as deep stall is likely to develop if prompt recovery action is not taken immediately a stall warning is received. In aircraft having this configuration the stall warning has to be more direct than a simple aural alert; initially the system applies vibration to the control column (stick shaking) and in some aircraft, if pilot response is not immediate, it is followed by a forward pressure to the column (stick pushing).

Angle of attack sensing

In general aviation aircraft the angle of attack sensing device often used consists of a vane mounted in the leading edge of a wing at the point where stagnation occurs at normal flight attitudes. Under these circumstances the vane occupies a neutral, mid-position with the air pressure approximately

equal on both its upper and lower surface. As the aircraft attitude becomes increasingly pitched nose-up, the stagnation point moves toward the lower surface of the wing and the pressure beneath the vane becomes significantly greater than that above it. The vane therefore moves upward and closes a switch attached to it, completing an electrical circuit to the stall warning horn in the cockpit. The mechanism is adjustable, so that it can be set to activate the stall warning at a precise angle of attack, slightly lower than the stalling angle. This type of sensor is illustrated in Figure 6.10.

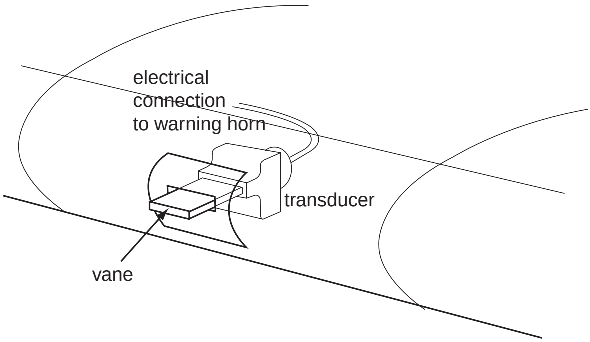

Figure 6.10 Leading edge angle of attack vane.

Some light aircraft and training types use an even simpler stall warning device known as a plenum chamber. An adjustable plate with a slot cut in it is mounted on a wing leading edge and adjusted so that the slot aligns with the stagnation point at normal flight attitudes. The slot is connected to a chamber, which is in turn connected by tube to a reed-operated horn in the cockpit. When the angle of attack reaches a preset value, the reduced pressure at the slot induces airflow through the horn, vibrating the reed to provide an aural alert to the pilot.

In larger aircraft, where the stall warning may well be accompanied by stick shaking or pushing, a more sophisticated type of angle of attack sensor is typically used. This uses an aerodynamic vane that projects into the airstream at a location unaffected by disturbances, usually on the side of the fuselage toward the nose of the aircraft. The vane is dynamically balanced and is connected to the rotor of a synchro, which transmits an electrical signal proportional to the angle taken by the vane, relative to a preset null position. Since the vane has a symmetrical aerofoil section and is freely balanced, it naturally aligns itself with the airstream when the aircraft is in flight. Thus, as aircraft pitch attitude changes, the vane remains aligned with the airstream and the aircraft moves in pitch relative to the vane. The relative movement is sensed by the vane synchro and transmitted to the stall warning system and the angle of attack indicator, where fitted. An angle of

attack vane is illustrated in Figure 6.11. Upon installation, the vane unit is accurately aligned by locating pins on the fuselage that engage in holes in the vane unit mounting flange.

Because the stalling angle of attack changes when flaps and slats are deployed, it is necessary to bias the vane synchro with inputs from the flap and slat position synchros to adjust the stall warning angle of attack accordingly.

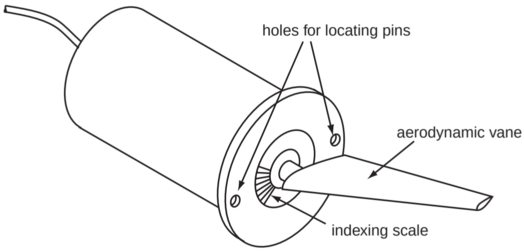

Figure 6.11 Angle of attack sensor unit.

Stick shaker

In aircraft where an immediate response to a stall warning is required it is usual to install a stick shaker. This consists of a d.c. motor that, when activated, applies vibration to the control column, immediately drawing the pilot's attention to the need to apply forward stick to reduce the pitch attitude of the aircraft. This method is chosen because it is reminiscent of the pilot's earliest flight training in stall recovery procedures in a training aircraft. The motor will continue to vibrate the control column until the angle of attack sensor senses that the aircraft's angle of attack has reduced below the warning threshold, whereupon the system cuts off d.c. supply to the motor. Figure 6.12 shows a block diagram of a stick shaker system. Switching the system control switch to TEST, with the aircraft on the ground, activates a small motor to rotate the dial of an indicator, proving that the control circuit is functional.

Stick pusher

Aircraft with poor stall recovery characteristics, such as T-tail configured aircraft, may be fitted with a system that actually pushes the control column forward if the pilot does not respond to the stick shaker. It was found with the BAC 1-11 and Trident aircraft, both high T-tailed with aft-mounted engines, that if stall recovery was not promptly executed, the aircraft was prone to enter a deep-stall (super stall) condition from which recovery is almost always impossible. The reason for this is that the tailplane (horizontal

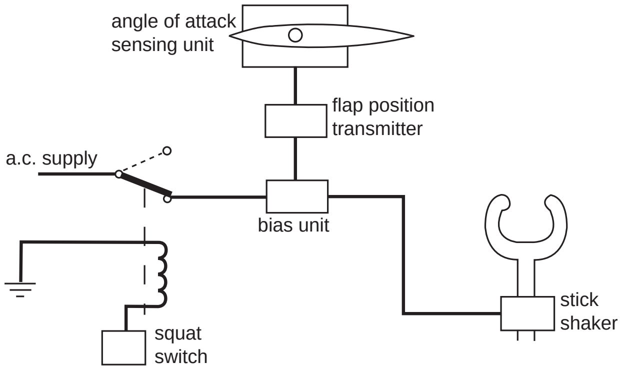

Figure 6.12 Stall warning system block diagram.

stabiliser) becomes stalled by the turbulent airflow from the stalled wings and all longitudinal control is lost. The aircraft enters a steep descent in a nose-up, wings level attitude. Thus, it is essential that rapid corrective action is taken in aircraft of this configuration, hence the stick pusher. The motive force for pushing the control column is often of the linear actuator type and the signal to it is passed through the elevator channel feel and centring unit. Activation of the stick pusher automatically disengages the automatic flight control system.

Flight data recorder

The Joint Aviation Authority (JAA) requires that all turbine powered aircraft with a take-off weight greater than 5700 kg and with seating for more than nine passengers shall be equipped with a flight data recorder. The device must be capable of retaining data recorded during at least the last 25 hours of aircraft operation, although this figure may be reduced to 10 hours for aircraft with a take-off weight of less than 5700 kg.

The data recorded must be sufficient to establish the following flight parameters:

- Altitude
- Airspeed
- Heading
- Attitude in pitch and roll
- Acceleration
- Thrust or power on each engine
- Configuration of lift or drag devices
- Radio transmission keying
- Use of automatic flight control systems

- Angle of attack
- Air temperature

For aircraft with a take-off weight in excess of 27 000 kg it is required that additional data must be recorded in order to be able to establish the following parameters, as well as those listed above:

- Primary flight control positions
- Pitch trim
- Primary navigation information as displayed to the flight crew
- Flight deck warnings
- Landing gear position
- Radio altitude

The data recorded must be from essentially the same sources as those which supply the information displayed to the flight crew and it must include any parameters that are peculiar to the operating characteristics of the aircraft design.

The flight data recorder must automatically begin recording all the above data before the aircraft is capable of moving under its own power and must automatically cease recording after the aircraft is no longer capable of moving under its own power. In practical terms, this usually means that recording starts with start of the first engine and ceases at shut-down of the last engine. The recorder must be contained within a container painted in a distinctive orange or yellow colour and its recovery must be assisted by reflective material and an underwater locating device that is automatically activated upon immersion.

It must be so installed in the aircraft that the probability of damage to the recorded data from shock, heat or fire is minimised. This is usually satisfied by locating the flight data recorder as far aft as practicable, typically in the vicinity of the rear pressure bulkhead.

The electrical supply to the recorder must be from a bus bar that can be expected to provide power under all circumstances, without jeopardising essential or emergency services. There must also be a pre-flight testing facility to check the serviceability of the recorder.

Figure 6.13 shows a block diagram of the typical inputs to the flight data recorder of a large transport aircraft.

Types of flight data recorder

Flight data recorders are classified according to the amount of information to be retained and the length of aircraft operating time over which data are to be recorded and stored. Recorders meeting the JAR-OPS requirements for

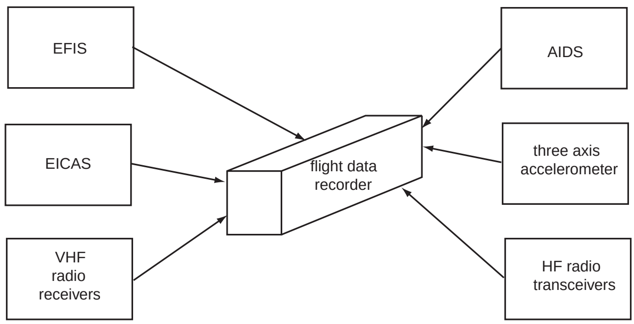

Figure 6.13 Flight data recorder block diagram.

aircraft with a maximum take-off weight in excess of 27 000 kg must be capable of recording at least 32 parameters and these are classified as Type I flight data recorders. Type II recorders meet the JAR-OPS requirements for smaller aircraft (take-off weight of 5700 kg) and these must be capable of recording at least 15 parameters. Type IIA recorders only have a 30-minute recording span, but must be capable of retaining data recorded during the preceding take-off.

The minimum 32 parameters required of Type I flight data recorders are listed below. Normally a Type II recorder would record the first 15 of these parameters, although the parameters may vary according to aircraft type.

- UTC or elapsed time
- Pressure altitude
- Indicated airspeed
- Heading
- Vertical acceleration
- Pitch attitude
- Roll attitude
- Radio transmission keying
- Power on each engine
- Trailing edge flap position or control selection
- Leading edge flap position or control selection
- Thrust reverser position
- Ground spoiler and/or speed brake position
- Outside air temperature
- Autopilot, autothrottle and automatic flight control system modes and status
- Longitudinal acceleration
- Lateral acceleration
- Primary control surface positions and/or pilot's control inputs

- Pitch trim setting
- Radio altitude
- Glide path deviation
- Localiser deviation
- Marker beacon transit
- Master warnings
- Navigational radio frequencies
- DME distances
- Landing gear status from squat switch
- Landing gear selector position
- GPWS
- Angle of attack
- Hydraulics systems pressures
- Latitude and longitude, groundspeed and drift angle

System monitoring

The flight data recorder system has its own built-in test equipment (BITE) and the serviceability of this and the recorder should be checked before the first flight of the day. FDRs are subject to annual inspection and to calibration on a 5-year cycle. Dedicated airspeed and altitude sensing equipment is subject to bi-annual inspection and calibration.

Aircraft integrated data system

Many of the larger transport aircraft types are equipped with data gathering and retention systems for monitoring the health and performance of the engines and aircraft systems. The system most commonly used is known as the aircraft integrated data system (AIDS), which provides the option of a real time display of current operating conditions, or downloading and print-out of the data when the aircraft is on the ground. Some operators make use of an extension to AIDS known as the aircraft communication addressing and reporting system (ACARS), whereby the system can be interrogated from the operator's ground base and technical data downloaded whilst the aircraft is in flight. The data recorded and stored by AIDS can be interchanged with the flight data recorder and the FDR data can be printed out during aircraft maintenance.

Cockpit voice recorder

The Joint Aviation Authority requires that all multi-engine turbine-powered aircraft with a maximum take-off weight in excess of 5700 kg and with seating for more than nine passengers shall be equipped with a cockpit voice

recorder. The voice recorder must be capable of retaining recorded information over the period of the last 2 hours of operation and the parameters recorded must be as follows:

- All radio voice communications received or transmitted from the flight deck.
- All sounds within the flight deck environment, including audio signals received by each boom and mask microphone in use.
- Voice communications between flight crew members on the interphone systems.
- All voice or audio signals identifying navigation or approach aids, as received on crew headphones or speakers.
- All announcements made by the flight crew on the public address system.

For aircraft with a maximum take-off weight of less than 5700 kg the recording time may be limited to 30 minutes.

The cockpit voice recorder must automatically begin recording before the aircraft first moves under its own power and continue until it is no longer capable of moving under its own power. In practical terms, this is usually from first engine start to last engine shut-down.

The voice recorder container must be easy to locate in a crash situation by painting it a distinctive orange or yellow colour with reflective material attached. It must also include an automatically activated underwater detection device and it must be resistant to shock, heat and fire.

The recorder must be installed in a location where its recordings are least likely to suffer damage. The site chosen is usually as far aft as practicable, typically close to the rear pressure bulkhead. It must receive its electrical power from a bus bar that can be relied upon to continue providing power under all circumstances and that is separate from the aircraft's essential and emergency services.

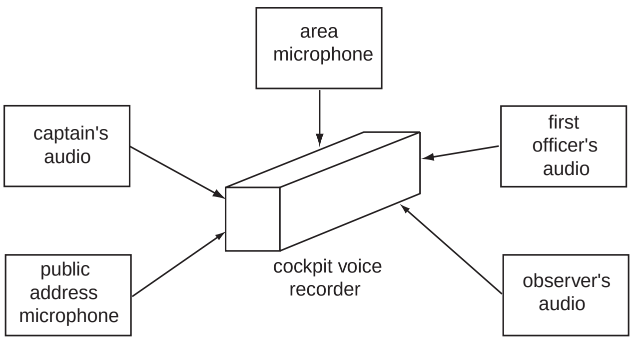

Figure 6.14 Cockpit voice recorder block diagram.

Figure 6.14 shows a block diagram of a cockpit voice recorder system for a large passenger transport aircraft.

There must be a means of preflight checking the cockpit voice recorder for serviceability. An aircraft may only be despatched with an unserviceable recorder provided that the means of repairing it are not available and that the aircraft does not complete more than eight subsequent consecutive flights with the device unserviceable.

Sample questions

1. The type of fly-by-wire system that converts pilot demands into rate of pitch or roll is known as:

   a. An active control system?
   b. A passive control system?
   c. A rate control system?
   d. An integral control system?

2. A yaw damping system is necessary in aircraft:

   a. To control the rate of yaw following a pilot rudder input?
   b. That are susceptible to Dutch roll?
   c. That are susceptible to phugoidal flight?
   d. That do not have swept wings?

3. An automatic stabiliser trim system:

   a. Adjusts the position of the elevators to align with the horizontal stabiliser?
   b. Adjusts a trim tab on the elevators to reduce elevator deflection?
   c. Adjusts the position of the horizontal stabiliser to centralise the elevators?
   d. Can only be engaged after the autopilot system has been engaged?

4. The altitude alert system will provide:

   a. An aural and visual alert that commences when the aircraft is within 900 ft of the selected altitude and continues until it is within 300 ft of the selected altitude?
   b. An aural and visual alert that commences when the aircraft is within 900 ft of the selected altitude and repeats when it is within 300 ft of the selected altitude?
   c. An aural alert that sounds for 2 seconds when the aircraft is within 900 ft of the selected altitude and a visual alert that remains illuminated until the aircraft is within 300 ft of the selected altitude?

d. An aural alert that sounds for 2 seconds when the aircraft is within 600 ft of the selected altitude and a visual alert that remains illuminated until the aircraft is within 300 ft of the selected altitude?

5. Radio altimeters operate in the .... band with a frequency range of ....:

a. UHF 4200 MHz–4400 MHz?
b. SHF 1600 MHz–1700 MHz?
c. UHF 4200 MHz–4400 MHz?
d. SHF 4200 MHz–4400 MHz?

6. The sweep rate of a radio altimeter is, typically:

a. High, to avoid interference with other aircraft transmissions?
b. Low, to avoid height ambiguity?
c. Low, because the transmitter and receiver are not co-located?
d. Variable, to make each aircraft's transmitted signals unique?

7. The accuracy of a radio altimeter is given as:

a. ± 1 ft or $\pm 5\%$ of the indicated height, whichever is the lesser?
b. ± 3 ft or $\pm 5\%$ of the indicated height, whichever is the greater?
c. ± 1 ft or $\pm 3\%$ of the indicated height, whichever is the greater?
d. ± 5 ft or $\pm 3\%$ of the indicated height, whichever is the lesser?

8. A basic GPWS requires inputs from (answer a, b, c or d):

1. Radio altimeter
2. Central air data computer
3. ILS glidepath receiver
4. Approach configuration
5. Navigation computer
6. Flight management system

a. 1, 2, 3, 4 only?
b. 1, 2, 3, 4, 5 only?
c. 1, 2, 3, 4, 5, 6?
d. 1, 2, 3 only?

9. GPWS mode 1 gives warning of:

a. Excessive descent rate?
b. Excessive terrain closure rate?
c. Altitude loss after take-off or go-around?
d. Unsafe terrain clearance?

10. The GPWS mode that gives warning of altitude loss after take-off or go-around is active between:

    a. 50 ft agl and 1800 ft agl?
    b. 50 ft agl and 700 ft agl?
    c. 50 ft agl and 2450 ft agl?
    d. 50 ft agl and 500 ft agl?

11. GPWS mode 5 is inhibited when:

    a. Mode 1 is active?
    b. The aircraft is below the glideslope?
    c. Mode 3 is active?
    d. Flaps and landing gear are deployed?

12. A fundamental difference between advanced GPWS and basic GPWS is that the former:

    a. Has only five operating modes?
    b. Gives alerts, rather than warnings?
    c. Operates from a greater height above ground level?
    d. Identifies the warning mode with an alert message?

13. The colour display generated by the enhanced GPWS shows terrain that is below the aircraft's projected flight path:

    a. In shaded yellow?
    b. In solid yellow?
    c. In shaded red?
    d. In green?

14. EGPWS mode 7 gives warning of:

    a. Windshear, below a radio altitude of 1500 ft?
    b. 'Minimums', below a radio altitude of 1000 ft?
    c. Terrain that is well above the projected flight path?
    d. Clear air turbulence?

15. TCAS II transponders transmit an interrogating signal on a frequency of .... and respond on a frequency of ....:

    a. 1090 MHz 1030 MHz?
    b. 4200 MHz 4400 MHz?
    c. 1030 MHz 1090 MHz?
    d. 1030 GHz 1090 GHz?

16. A TCAS II message that relates to a possible confliction and requires avoiding action is known as a:

    a. Traffic advisory?
    b. Resolution advisory?
    c. Manoeuvring advisory?
    a. Vertical speed advisory?

17. A TCAS II symbol depicting a solid yellow circle indicates:

    a. No threat?
    b. Proximate traffic?
    c. Traffic advisory?
    d. Resolution advisory?

18. The minimum altitude at which the TCAS II system will issue a traffic advisory is:

    a. 400 ft?
    b. 500 ft?
    c. 1000 ft?
    d. 1800 ft?

19. The aural warning usually associated with the overspeed warning is:

    a. A chime alert?
    b. A gong?
    c. A warbling tone?
    d. A clacker?

20. A stall warning system is set to operate an alarm:

    a. At a speed just below stalling speed?
    b. At an angle of attack just below stalling angle?
    c. At a speed just above stalling speed?
    d. At an angle of attack just above stalling angle?

21. The minimum retained data period required by JAR-OPS for a flight data recorder installed in an aircraft with a take-off weight in excess of 5700 kg and seating for more than 9 passengers is:

    a. 25 hours?
    b. 30 minutes?
    c. 10 hours?
    d. 30 hours?

22. The minimum number of parameters required to be covered by a Type I FDR is:

    a. 25?
    b. 32?
    c. 15?
    d. 10?

23. A cockpit voice recorder must automatically begin and cease operating:

    a. At take-off and touchdown?
    b. During voice transmissions by the flight crew?
    c. Before the aircraft first moves under its own power and after it is no longer capable of moving under its own power?
    d. From start of take-off roll to end of landing roll?

Chapter 7
Powerplant and System Monitoring Instruments

In Chapter 4 we saw how engine and system monitoring is presented to the pilots by electronic displays such as EICAS and ECAM. In this chapter we shall be examining the type of instruments often found in less sophisticated aircraft, and the methods of measurement of pressure, temperature, fluid flow and quantity, rotary speed, torque and vibration used in all aircraft.

Pressure gauge

In aircraft of British and American manufacture it has been conventional to measure pressure in units of pounds per square inch (lb/in^2 or psi), whereas most European manufacturers preferred metric units and used kilograms per square centimetre (kg/cm^2). Recently the trend in European manufactured aircraft has been to measure pressure in atmospheric units known as bars, one bar being equal to 14.7 psi. Low pressures, especially where related to atmospheric pressure, are usually measured against absolute zero in inches of mercury (in Hg).

The method of pressure measurement depends largely upon the value of the pressure to be measured and how it is to be displayed. Not surprisingly, high pressures such as those found in a hydraulic system, for example, require more robust methods than the low pressures associated with piston engine manifolds. In early aircraft it was normal to connect the pressure instrument (gauge) on the pilot's instrument panel direct to the pressure source, in which case the pressure measuring device is contained within the instrument itself. Thus, for example, engine oil pressure is piped to a flexible element within the pressure gauge and this is known as a direct-reading instrument. The disadvantages of such a system are that a leak in the connecting pipe not only renders the pressure gauge useless, but also presents a loss of vital engine lubricant and a potential fire hazard. Furthermore, the weight of piping required to convey high pressure fluids is not inconsiderable.

To overcome these disadvantages it has become the usual practice in all but the simplest cases to use remote reading instruments, in which the

pressure is measured at source and transmitted to the pilot's instruments by mechanical or electrical means. A simple mechanical transmitter, now rarely used, consisted of a cylinder containing a free piston. One side of the piston was connected to the pressure source, say hydraulic system pressure, and the other was connected to the instrument panel pressure gauge by an enclosed pipe filled with non-flammable fluid. The force exerted on the piston by the hydraulic system pressure was transmitted by the enclosed system to the pressure gauge. Such methods have been replaced almost entirely by electrical transmission of a signal proportional to the pressure exerted on a transducer, the signal then operating a suitably calibrated indicator in the cockpit.

Direct-reading pressure gauges

Where relatively high pressures are involved, such as engine oil pressure or hydraulic systems, the type of pressure measuring element commonly used in direct-reading pressure gauges is the Bourdon tube, as illustrated in Figure 7.1. The device comprises a flattened tube formed into a semi-circle and closed at one end. The other end of the tube is connected to the pressure source. The tube is made of a flexible material so that, when pressure is applied to the inside of it, it tends to straighten. This tendency is opposed by a spring, or by the natural resistance of the tube itself, so the extent of the 'straightening' movement is proportional to the pressure applied. The closed end of the tube is connected through gearing to the instrument pointer, which moves against a calibrated scale to indicate system pressure in the chosen units of measurement.

Where lower pressures are to be measured the Bourdon tube is not sufficiently sensitive. The pressure measuring elements used are typically corrugated capsules of the type described in Chapter 1 under air data instruments. Figure 7.2 illustrates an example of the use of these elements in a piston engine manifold pressure gauge. Manifold air pressure (MAP)

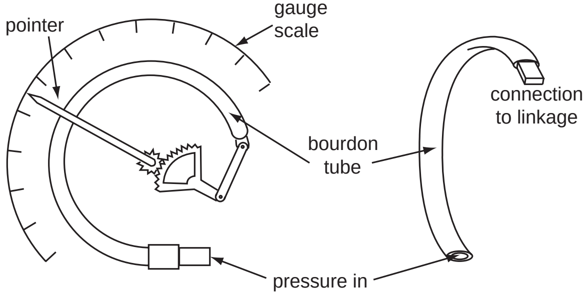

Figure 7.1 Bourdon tube principle.

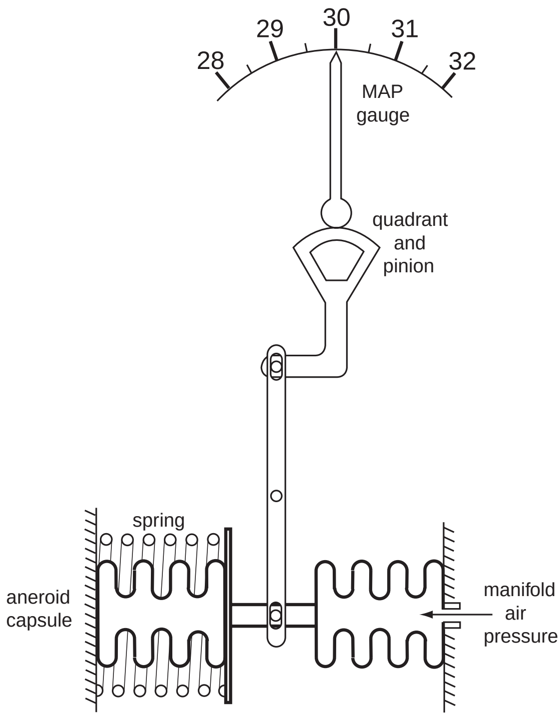

Figure 7.2 Manifold air pressure gauge operating principle.

typically ranges from a value less than ambient atmospheric pressure to a small amount (perhaps 1 or 2 bar) above, and so it must be measured against absolute zero (i.e. the pressure in a total vacuum). To achieve this, two sensing capsules are used, one of which is evacuated and spring loaded to respond to ambient atmospheric pressure and the other of which is connected internally to the engine manifold by piping. The two capsules are linked mechanically to the instrument pointer.

The manifold air pressure or boost gauge is usually calibrated to read absolute pressure in inches of mercury (in Hg), thus at sea level with the engine stopped it will indicate approximately 30 in Hg. When the engine is running and the pistons are drawing air into the cylinders through the intake manifold, a partial vacuum is created in the manifold and the gauge will read less than ambient atmospheric pressure. If the engine is supercharged, or 'boosted', the supercharger or turbocharger will force air into the manifold at higher engine powers and the manifold pressure will be greater than ambient atmospheric pressure. The MAP is a measure of the power being developed by a piston engine, hence the reason for indicating it to the pilot.

In the system illustrated in Figure 7.2 the situation depicted is that which would exist at sea level with the engine stopped. Ambient atmospheric

pressure is acting on the outside of the aneroid capsule, against the force of its internal spring, and on the inside and outside of the manifold capsule. The forces exerted are in balance and the gauge pointer is indicating 30 in Hg against the calibrated scale. If the engine were to be started and run at low rpm, the pistons would draw a partial vacuum in the intake manifold and the force exerted by the manifold capsule would decrease. This is because the ambient atmospheric pressure acting on the outside of the manifold capsule tends to compress the capsule and this is transmitted to the gauge pointer through mechanical linkage and gearing. The amount of compression is restricted by the atmospheric pressure acting on the outside of the sealed, aneroid capsule and so the pointer movement is proportional to the change in manifold air pressure.

At increased engine rpm and power the supercharger-boosted pressure in the manifold becomes greater than atmospheric pressure and the manifold capsule expands, moving the gauge pointer toward a higher value. The extent of movement is limited by the opposing force of the spring surrounding the aneroid capsule.

Remote-reading pressure gauges

The use of direct-reading pressure gauges is mainly restricted to small aircraft with a limited number of gauging requirements. The more complex the aircraft and its systems, the greater the number of pressure measurements required and the less practical it becomes to pipe these to an instrument array in the cockpit. Consequently, the various pressures are measured at source and transmitted electrically to the pilot's instrument displays, which may comprise individual electrically operated indicators or a computerised electronic display.

The conversion of pressure into a proportional electrical signal and its transmission to a calibrated indicating instrument necessarily involves the conversion of mechanical movement into an electrical output at the measuring source and, in the case of a mechanical indicator on the flight deck, a reversal of this conversion. There are various types of device for achieving this, including the synchronous transmission, or synchro, system, the inductive transmitter and the potentiometer system.

Synchronous transmission

The principle of synchronous transmission was described in Chapter 2 under the direct reading compass, but is repeated here as it applies to the measurement and transmission of engine oil pressure, and as illustrated in Figure 7.3.

In the example, oil pressure is sensed in a capsule that expands against a spring to create linear movement proportional to the measured pressure.

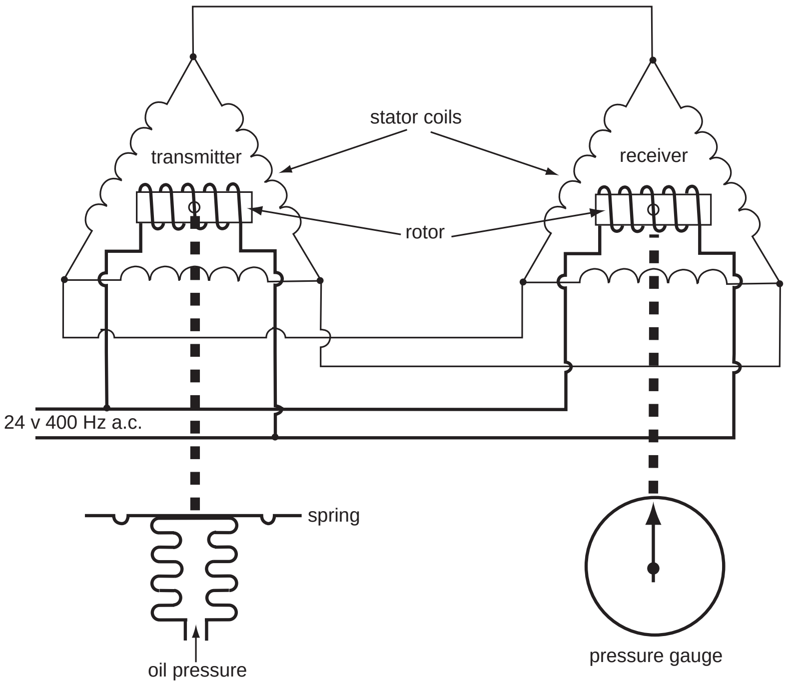

Figure 7.3 Synchronous transmission system.

This movement is transmitted mechanically to a rotor upon which is wound a coil carrying alternating current. The rotor is positioned centrally within a stator having three coils at 120° spacing. The electro-magnetic field created around the rotor induces a current flow in each of the stator coils, the strength of the current in each coil being dependent upon the orientation of the rotor. These three transmitter currents are fed to, and repeated in, an identical receiver stator system located behind the cockpit instrument panel, where they create a magnetic field identical to that in the transmitter. This field interacts with the a.c. induced field surrounding the receiver rotor coil, causing the receiver rotor to rotate and adopt an orientation corresponding to that of the transmitter rotor. The receiver rotor is mechanically connected to the pointer of a pressure-indicating instrument.

Induction transmitter

In this type of pressure transmission system the pressure to be measured is led to a capsule inside the pressure transmitter, which is located as close as possible to the pressure source. The capsule is mechanically connected to a permanent magnet armature, and linear expansion or contraction of the capsule moves the armature linearly against the opposition of a spring. The armature is surrounded by two sets of coils, supplied with current and

connected to a moving coil indicator on the flight deck. As the armature moves, its position relative to each of the coils differs, and the inductance of the two coils will vary in direct proportion to the pressure being measured. This will cause the output current from the coils to vary, positioning the pointer of the moving coil indicator pressure gauge accordingly. The principle is illustrated in Figure 7.4.

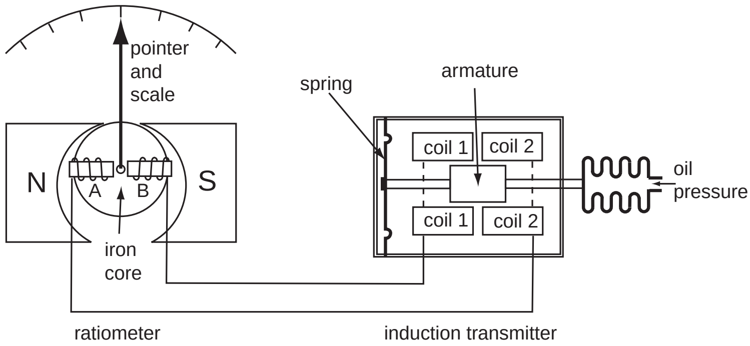

Figure 7.4 Induction transmitter and ratiometer.

The type of moving coil indicator typically associated with this type of remote reading pressure gauge is the ratiometer, which is illustrated in Figure 7.4 above. Current from the transmitter coils is supplied to two coils wound around armatures on a spindle-mounted iron core. The core is positioned eccentrically between the poles of a permanent magnet, so that the air gap between the core and the magnet poles is greater on one side than on the other. Where the gap is greatest, the strength of the permanent magnetic field will be weakest and vice versa. Current flowing through the core coils creates electro-magnetic fields that will interact with the permanent magnetic field. If the two current flows are equal, their magnetic field strengths will be equal and of opposite polarity, thus cancelling each other and the iron core will be held stationary. If, however, the current flow in coil B is greater than that in coil A the stronger magnetic field surrounding coil B will be attracted toward the larger air gap where the field strength is weaker, rotating the iron core on its spindle and moving the attached gauge pointer against a calibrated scale. At the same time, the weaker induced field surrounding coil A is moved into a narrowing air gap where the permanent magnetic field is stronger. This will eventually arrest the rotation of the core when the opposition of the permanent field matches the attraction of the induced field and the gauge pointer will indicate the changed pressure that caused the current imbalance. If the current flow in coil A is greater than that in coil B, the effect will be the reverse of that described above.

Potentiometer transmission

This type of pressure transmission system uses an inductance transmitter similar or identical to that described above, but its output current is amplified and used to drive an a.c. motor, which is connected to the pointer of the pressure gauge and a potentiometer. In simplistic terms, the potentiometer is a variable resistance connected to an a.c. supply, the output of which is fed back to the transmitter amplifier. As the motor drives the gauge pointer, the potentiometer resistance varies until its output current and phase balances the transmitter signal and supply to the motor ceases, holding the gauge pointer at its new position.

Piezo-electric transmitters

In most large modern aircraft the transmission of low pressure utilises solid-state transmitters that operate on the piezo-electric principle. These comprise a thin stack of quartz discs impregnated with metallic deposits. When acted upon by pressure the disc stack flexes and small electrical charges are produced. The polarity of the induced charge depends upon the direction of flexing, due to increased or decreased pressure, and the output is amplified and used to actuate an electronic representation of a pressure indicator.

Pressure gauge indications

Traditionally, aircraft pressure gauges use a central pointer moving against a circular calibrated scale, as illustrated in Figure 7.5. The gauge scale is calibrated in the chosen units of pressure measurement and coloured markings are added in many cases to indicate operating limits

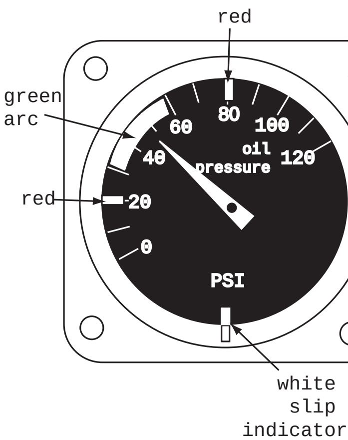

Figure 7.5 Typical piston engine oil pressure gauge.

and ranges. In the case of the engine oil pressure gauge illustrated, maximum and minimum oil pressures are indicated by radial red lines and the normal operating range of pressures by a green arc. If these markings are made on the glass cover, rather than the instrument face, it is required that a white radial line, known as a slip indicator, must be painted on the glass and the adjoining casing to indicate movement of the glass relative to the casing.

Piston engine manifold air pressure (MAP) gauges are also usually colour coded to indicate operating power ranges and limits. A red radial line indicates the maximum permissible MAP for take-off power and blue and green arcs indicate the lean and rich mixture ranges, respectively. Normally aspirated (unsupercharged) piston engines with fuel injection systems often have a fuel pressure gauge with coloured arcs to indicate lean and rich mixture ranges, since in these powerplants fuel pressure is directly proportional to engine power. In the cruise at reduced power a lean mixture may be used for fuel economy, but at high power settings it is essential to use a rich mixture to avoid detonation and engine damage. The coloured arcs are the same as those on a MAP gauge.

Pressure operated switches

In many cases the pilot does not need to know the actual operating pressure of a particular system, but merely that it is within acceptable limits. Examples of this are the constant speed drive unit (CSDU) oil pressure, where only a low pressure warning is necessary, and similarly the hydraulic system pressure in some light aircraft with very limited hydraulically operated devices. In these cases it is usual to use a simple transmitter in the form of a pressure-operated switch connected to a warning light in the cockpit. The source pressure is applied to a small piston in the switch assembly, the movement of which is opposed by a calibrated spring. When the source pressure is within operating limits the spring is compressed and a switch connected to the piston is held with its contacts open. If pressure falls below a preset value, the spring overcomes the pressure acting on the piston and moves the switch to close the contacts and connect supply current to the warning light. Clearly, the same principle, but with suitably amended calibration, can be adapted to indicate excessive pressure. An example of this is the oil filter bypass warning light, which will be activated as the filter becomes clogged and the pressure differential across it increases, to warn the pilot that the filter bypass valve will open unless the filter is changed at the earliest opportunity.

Temperature gauge

Temperature gauges are used in aircraft piston engines to monitor lubricating oil temperature and cylinder head temperature (CHT). Additionally, carburettor air intake temperature is measured in some engines, to give warning of carburettor icing, and exhaust gas temperature may also be monitored, since this gives an indication of combustion efficiency and is useful when adjusting mixture settings. In aircraft turbine engines lubricating oil temperature and exhaust gas temperature (EGT) are invariably monitored. In the case of turbo-propeller engines, turbine inlet gas temperature may be measured instead of EGT.

There are fundamentally two methods of temperature sensing in common use in aircraft engines and systems: the variable resistance method and the thermocouple. The variable resistance type of sensing element makes use of the tendency of metals to change their conductivity as temperature changes, such that the conductivity decreases (i.e. resistance increases) with increasing temperature. The thermocouple operates on the principle that heat energy can be converted into electrical energy due to what is known as the Seebeck effect. The type of sensor used depends mainly upon the degree of heat involved. The sensing elements that make use of resistive change with temperature are not generally suitable for use with the high temperatures associated with exhaust gas, but are ideal for the temperature ranges experienced in engine lubricating oil systems. The electrical energy generated by the Seebeck effect is small, so this type of sensor is better suited for measurement of high temperatures.

Resistive systems

A resistive temperature sensing system comprises a sensing element containing a resistance element supplied with low voltage electrical current and connected in series with an indicator that will convert the electrical output of the sensor into mechanical movement of an instrument pointer. The electrical supply is usually d.c., but in some cases single phase a.c. may be used. The sensor element is contained within a closed tube, or 'bulb', which is immersed in the fluid to be measured. As fluid temperature increases, the resistance of the element will increase and current flow to the indicator unit will decrease in proportion. A typical sensing unit is illustrated in Figure 7.6.

Sensing unit

The d.c. supply is led to the sensing probe through a two-pin connector. The resistive element is wound around a central core made of non-conducting material and enclosed within a leakproof casing made of thin, heat-conducting material, typically copper or aluminium. The tube is inserted

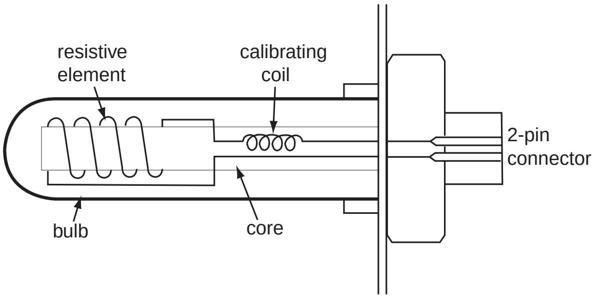

Figure 7.6 Resistive temperature sensing unit.

into the fluid system (e.g. the engine lubricating oil system) and held in place by a threaded union nut. The calibrating coil shown in Figure 7.6 has a resistance value that is set during manufacture to determine the temperature/resistance characteristic of the sensing probe for the temperature range it is intended to measure.

Indicating unit

The temperature gauge associated with resistive temperature measurement is typically a moving coil instrument, which may be operated by a Wheatstone bridge circuit. The principle of operation of the Wheatstone bridge is illustrated in Figure 7.7.

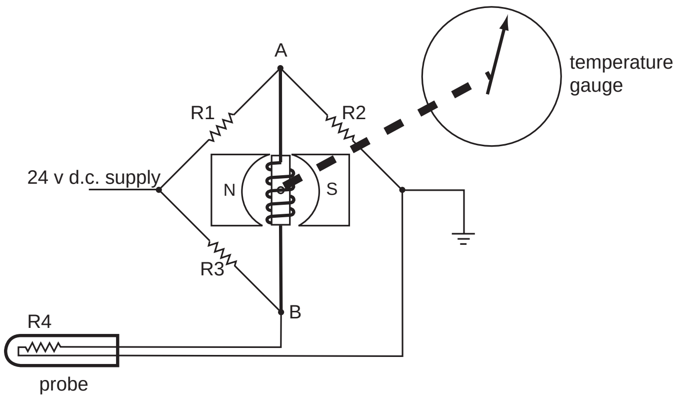

Figure 7.7 Temperature probe and Wheatstone bridge circuit.

The Wheatstone bridge comprises two pairs of series resistances, connected in parallel. The resistances are connected to a low voltage circuit, typically 24 V d.c. In the diagram above, resistances R_1, R_2 and R_3 are of identical value, whilst resistance R_4 is the element of the temperature probe

and varies with the temperature of the sensed fluid. Resistances R_1 and R_2 are connected in series and form one side of the bridge system, whilst resistances R_3 and R_4 are also connected in series and form the other side. A coil surrounding an iron core is connected from point A, between resistances R_1 and R_2, to point B between resistances R_3 and R_4.

Let us assume for the moment that the temperature of the sensed fluid is such that the resistance value of R_4 is the same as the other three resistances. Since the total resistance on each side of the parallel bridge circuit is the same, it follows that the current flow will also be the same on each side. Bearing Ohm's Law in mind, it therefore follows that the voltage at points A and B will be identical and there will consequently be no current flow through the armature coil, since current will only flow from a higher voltage point to a lower one.

Suppose now the temperature of the sensed fluid increases. The resistance of the probe element will increase and current flow through resistance R_4 will decrease (Ohm's Law again, $I = V/R$), whilst the current flow through resistance R_2 remains constant. As a result, the voltage at point B will increase ($V = IR$), whilst the voltage at point A remains constant; the voltage difference will cause a current flow from B to A and this current flow will induce a magnetic field about the coil, concentrated in the soft iron armature. The armature is situated within a permanent magnetic field, and magnetic attraction/repulsion will cause the armature to rotate upon its spindle. The direction of rotation will depend upon the polarity of the armature field, which is in turn dependent upon the direction of current flow in the armature coil. The armature is connected mechanically to the pointer of the temperature gauge.

If the temperature of the sensed fluid were to decrease, the resistance of the probe element would decrease and current flow through resistance R_4 would increase above that through resistance R_2. The voltage at point A would now be greater than that at point B and current flow through the armature coil would be from A to B, reversing the polarity of the induced magnetic field. The armature would consequently rotate in the opposite direction, indicating the reduced temperature on the gauge.

If you find this concept hard to understand, consider the analogy of water flowing through a similar canal system, as shown in Figure 7.8. If the flow is restricted at R_4, it is clear that it must divert through the path provided by the interconnecting ditch. If the restriction at R_2 and R_4 is the same, there will be no flow through the interconnecting ditch, and so forth.

Thermocouple sensors

Thermocouple temperature measuring sensors require no external electrical supply, since they directly convert heat energy into electrical energy. They

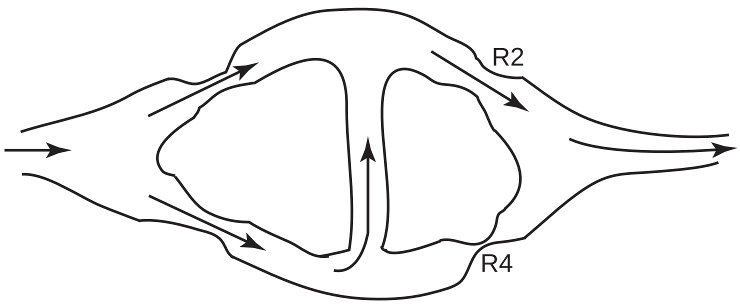

Figure 7.8 Wheatstone bridge principle.

operate on the principle that, if two conductors made of dissimilar metals are connected at either end, as illustrated in Figure 7.9, a potential difference will exist between the two junctions provided that there is a temperature difference between the junctions. The value of the potential difference will be directly proportional to the temperature difference and, since the two joined conductors form a loop, will cause current to flow around the loop. Clearly, the greater the temperature/potential difference the greater the current flow and, when suitably amplified, the thermocouple current can be made to operate the indicator of a temperature gauge.

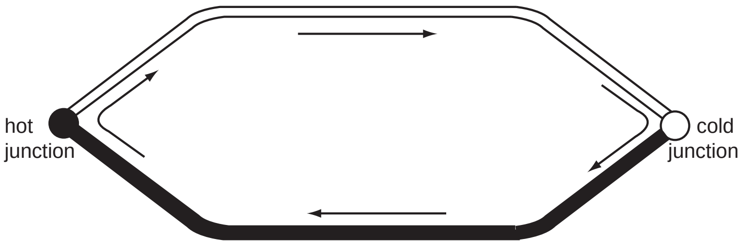

Figure 7.9 Thermocouple principle.

For the measurement of piston engine cylinder head temperature, where the temperature range is typically of the order of 400°C to 850°C, the metals used are usually copper and copper–nickel alloy, or iron and copper–nickel alloy for the higher end of the range. For gas turbine exhaust gas temperature measurement, where the maximum temperature may be as high as 1100°C, nickel–aluminium and nickel–chromium alloy conductors are usually used.

Cylinder head temperature thermocouples typically take the form of a 'washer' bolted to the cylinder head and forming the hot junction of the thermocouple. The cold junction is at the amplifier of the temperature

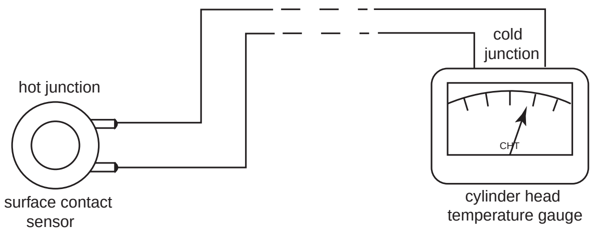

Figure 7.10 Surface contact sensor.

indicator. This is known as a surface contact sensor and is illustrated in Figure 7.10.

The measurement of gas temperature uses an immersion sensor which, as its name suggests, consists of a probe immersed in the hot gas flow, containing the hot junction of the thermocouple. The cold junction is at the indicator as before. An immersion probe is illustrated in Figure 7.11. In all cases the indicator contains a compensating device that automatically allows for variations in temperature at the indicator.

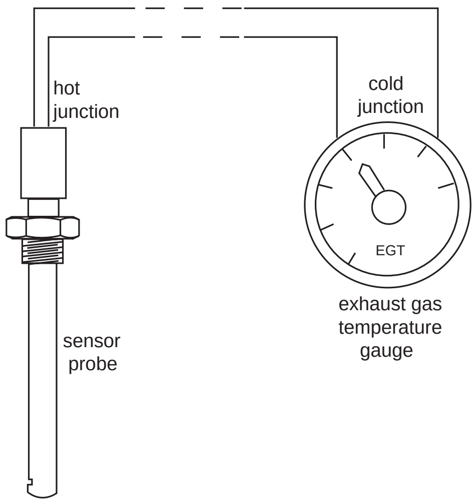

Figure 7.11 Immersion sensor.

Turbine exhaust gas temperature is usually measured within the jet pipe as close as possible to the turbine outlet. In order to allow for the harsh conditions, in which probes might become damaged, it is usual for a number of probes to be connected in parallel and positioned radially at intervals around the perimeter of the jet pipe.

Air temperature measurement

The measurement of intake air temperature is important in piston engines to provide warning of potential carburettor icing and in piston and gas turbine engines for the determination of engine performance. In the case of gas turbine performance, it is preferable that the measured temperature should be the static air temperature (SAT) at the altitude at which the aircraft is operating. However, as was discussed in Chapter 1 under air data computers, SAT cannot be directly measured because heating occurs due to compression and friction. The increased temperature due to this heating is known as the ram rise and the resultant temperature as total air temperature (TAT). The ram rise can be calculated and the air temperature gauge reading can be corrected, either automatically or from a chart, to give SAT. The ability of a temperature sensor to measure the full extent of the ram rise effect is known as its recovery factor and, in most modern sensors, is close to unity.

Meaning of coloured arcs

Temperature gauges, especially those used in conjunction with piston engines, usually have coloured arcs and radial lines to indicate operating temperature ranges and limits.

Engine oil temperature
The piston engine lubricating oil temperature gauge is usually marked with a green arc, typically between 60°C and 70°C, to indicate the normal operating temperature range. Red radial lines indicate minimum and maximum safe operating temperatures and these are typically 40°C and 100°C respectively.

Carburettor intake temperature
The carburettor air intake temperature gauge typically has three arcs coloured yellow, green and red. The yellow arc extends from −10°C to +15°C and indicates the temperature range within which a carburettor icing hazard exists. The green arc extends from +15°C to +40°C and indicates the normal operating temperature range. The red arc begins at +40°C and indicates intake temperatures that are liable to cause detonation within the cylinders.

Exhaust gas temperature
Exhaust gas temperature gauges are usually marked with red coloured arcs or a red radial line to indicate maximum temperature ranges or limits.

Vapour pressure gauge

A few light aircraft still use a very simple form of temperature gauge that operates on the Bourdon tube principle. The gauge is connected by capillary tube to a bulb filled with a highly volatile liquid. The bulb is immersed in the medium to be sensed (e.g. the engine oil system) and the heat of the medium vaporises the liquid in the bulb. The pressure in the closed system of bulb and tube increases in direct proportion to the medium temperature and acts upon the gauge Bourdon tube to move a pointer against a scale graduated to indicate temperature.

RPM indicator

The measurement of engine revolutions per minute (rpm) is important in unsupercharged piston engines, since it is an indication of the power being delivered to the propeller. Similarly, in gas turbine engines, rpm is related to thrust, although it is more common to measure this in terms of engine pressure ratio (EPR). In early single-engine aircraft the pilot's rpm indicator, or tachometer as it is properly known, was usually driven directly from the engine by means of a flexible drive and a system of flyweights. As aircraft became more complex this method became impractical and electrical transmission of the measured rpm to the pilot's instruments was developed.

Electrical tachometer

The method commonly used to achieve electrical transmission of rpm in piston engines is illustrated in Figure 7.12. A small three-phase a.c. generator is driven from the engine accessories gearbox and its output is used to drive a synchronous motor, which operates the pilot's engine tachometer. The a.c. frequency of the generator output will vary directly with engine rpm, and it is the supply frequency that determines the speed of rotation of an a.c. synchronous motor. Hence, the higher the engine rpm, the greater the rotary speed of the tachometer motor.

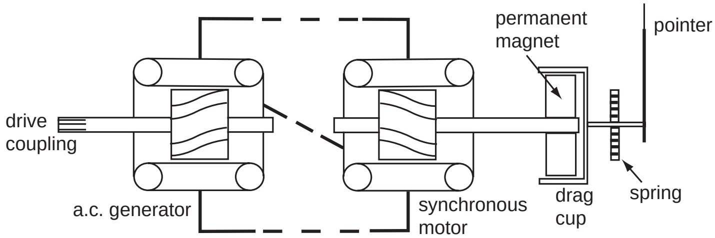

Figure 7.12 Electrical tachometer system.

The tachometer indicator is usually a conventional pointer moving through an arc against a calibrated scale, so it is clear that the continuous rotary motion of the synchronous motor must be converted into semi-rotary movement of the tachometer pointer. This is achieved by means of a magnetic device called a drag cup. Mounted on the rotor shaft of the synchronous motor is a permanent magnet which rotates inside an aluminium cup. The rotating field of the permanent magnet sets up eddy currents in the aluminium drag cup. These create electro-magnetic forces that react with the rotating permanent magnetic field and create a rotary force on the drag cup. The drag cup rotary movement is restrained by a coil spring attached to the shaft connecting it to the tachometer pointer. Thus, its amount of deflection is dependent upon the electro-magnetic force acting on the drag cup, which is in turn dependent upon the speed of rotation of the permanent magnet. The degree of pointer movement is therefore directly proportional to the engine rpm.

A typical piston engine rpm indicator is shown in Figure 7.13. It will be noted that the actual engine rpm is indicated, as is normal with piston engines. Gas turbine engine tachometers usually indicate rpm as a percentage, where 100% rpm is the optimum engine rotary speed. In the case of piston engine tachometers, a green coloured arc may indicate the normal operating rpm range, with a red radial line or arc to indicate maximum permissible rpm and time-limited operating rpm ranges.

Electronic tachometer

A type of electronic tachometer sometimes used with piston engines converts the impulses from the engine magneto into voltage, to drive an

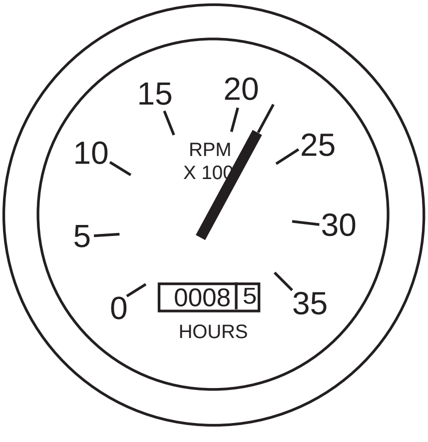

Figure 7.13 Typical piston engine rpm indicator.

indicator pointer. Clearly, the higher the engine rpm, the more impulses per minute from the magneto and the higher the voltage from the conversion circuit. When supplied to a voltmeter calibrated to read rpm, the amount of pointer deflection will be directly proportional to engine rpm. The advantage of both this type of tachometer and the electrical type is that they require no external electrical supply and will continue to operate in the event of failure of normal aircraft electrical services. There is a more complex type of magneto-driven electronic tachometer that requires a transistorised amplifier circuit needing a 12 V supply from the aircraft electrical system.

Servo-operated tachometer

Some gas turbine engine tachometers use a variation of the electrical tachometer described above, in which the generator output is converted into a square waveform by a solid-state circuit. A 'square' pulse is formed each half-cycle of the generator output, resulting in a pulse repetition frequency that is twice the a.c. generator output frequency. The pulsed transmission produces direct current (d.c.) to drive a d.c. motor, which operates the tachometer indicator pointer. The d.c. voltage, and therefore the motor speed, is dependent upon the pulse repetition frequency, which is in turn dependent upon engine rpm. This system is not independent of the aircraft electrical system, since it includes an overspeed pointer mechanism, which requires an external 28 V d.c. supply to reset it.

Tacho-probe system

This method of rpm measurement, illustrated in Figure 7.14, is commonly used with gas turbine engines as it has a number of significant advantages. Not only is it electrically independent, but its output can be used to supply flight data and autothrottle systems as well as to operate the rpm indicator.

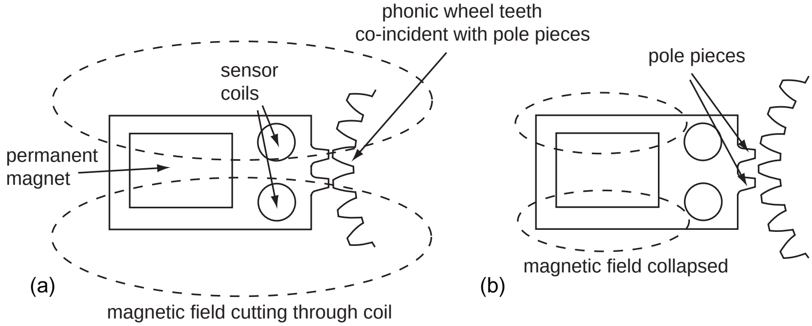

Figure 7.14 Tacho-probe system.

A toothed wheel is mounted on a shaft, the rpm of which is to be measured. In gas turbine engines this is usually the HP compressor/turbine shaft, but in many turbo-fan engines the fan rpm is also measured. This wheel is known as a phonic wheel and it clearly rotates at the same speed as the HP shaft or fan shaft. Mounted on the engine casing adjacent to the phonic wheel is a probe unit comprising a permanent magnet, two pole pieces that are spaced to exactly match the spacing of the phonic wheel teeth and sensing coils in which electrical current is generated.

When two of the phonic wheel teeth are exactly opposite the two pole pieces of the probe the permanent magnetic field surrounding the coils is at maximum strength, as in Figure 7.14(a). As the wheel rotates and the teeth are no longer co-incident with the pole pieces, as in Figure 7.14(b), the magnetic field strength surrounding the coils falls to near zero. This fluctuating field strength through the coil windings induces voltage in the coils and an alternating current flows through the associated output circuit. The frequency of the induced a.c. is directly proportional to the rotary speed of the phonic wheel and is used to actuate the rpm indicator pointer to show shaft speed as a percentage.

A typical gas turbine rpm indicator is shown in Figure 7.15. The large pointer shows percentage rpm from 0% to 100% in 10% increments. The smaller scale and pointer shows percentage rpm increments in unitary values from 0% to 10%. In the display illustrated the gauge is indicating 94% of optimum rpm.

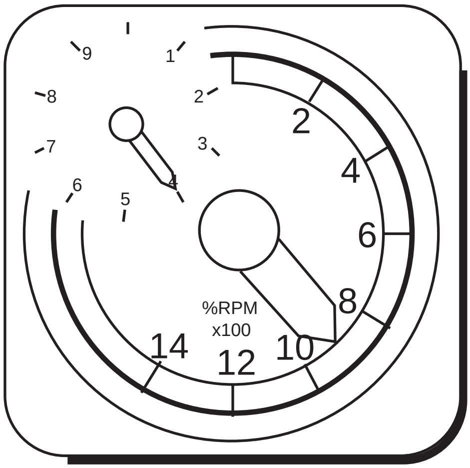

Figure 7.15 Typical gas turbine tachometer.

Flight hour meter

Figure 7.13 shows an hour meter incorporated in the piston engine tachometer display. It is important for routine servicing, inspection and component life to have a record of the number of flight hours accumulated by the aircraft. In light general aviation aircraft this is often achieved by a meter activated only when the engine is operating at cruise rpm. For example, if the normal cruise rpm is 2200 it follows that the engine will complete 132 000 revolutions in one hour at cruising speed. Thus, for each 132 000 revolutions the flight hour meter progresses by one unit. Such a meter suffers from the obvious limitation that it does not record flight times at other than cruise rpm. However, for aircraft that operate at this speed for all but take-off and landing, this is adequate and has the advantage of being independent of aircraft electrical power.

A slightly more sophisticated system is operated by an electric clock, powered from the aircraft electrical system and activated by the landing gear 'weight-on' or 'squat' switch. This system truly records flight time, since the switch supplying power to the clock only closes when aircraft weight is off the landing gear *and* the battery master switch is closed. This type of flight hour recorder is known as the Hobbs meter.

Fuel consumption gauge

It is important for the flight crew to be aware of the rate at which fuel is being consumed in flight in order to calculate range, endurance and economy. This information is vital to the operation of automated thrust and flight control systems. In large gas turbine powered transport aircraft the measurement of fuel flow is made by relatively complex flow metering systems that are capable of integrating the flow rate with time to compute and display both rate of fuel flow and the total fuel consumed. In smaller, short range transports, particularly those powered by piston engines, it is usual to measure only fuel flow rate using less sophisticated devices.

Fuel flow is ideally measured in terms of mass flow rather than volumetric flow, since it is the mass of fuel consumed that determines the power output of an engine. Mass per unit volume varies with the density of the fuel, which in turn varies with its temperature. Thus, if fuel flow is measured in terms of volumetric rate (gallons or litres per hour) a further calculation is necessary, taking temperature into account, in order to determine the mass flow rate. In short range aircraft this is less important, but in long range transports the flowmeter calibration usually takes fuel temperature into account and computes mass flow rates. Mass flow is usually measured in pounds (lb) or kilograms (kg) per hour.

Fuel flow and pressure

With a continuous flow fuel-injected piston engine the rate of fuel flow to the injectors is proportional to the fuel supply pressure. In piston-engine aircraft that are equipped with direct fuel injection systems it is not uncommon for the fuel pressure gauge to be calibrated to indicate fuel flow rate as well as fuel pressure. Since such aircraft are not usually long range types it is adequate for the fuel flow indication to be volumetric, in gallons per hour or litres per hour. An example of a fuel pressure gauge calibrated to serve also as a flowmeter is shown in Figure 7.16. It will be noted that the calibration indicates the fuel flow rates for various power settings (e.g. cruise, take-off and climb).

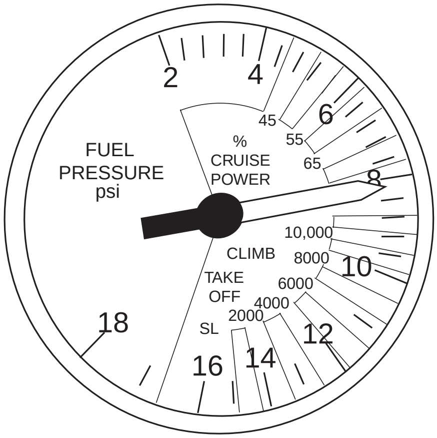

Figure 7.16 Fuel pressure gauge/flowmeter.

Vane-type flowmeter

The principal disadvantage of the combined pressure gauge/flowmeter is that it will provide erroneous indications if the fuel pressure is artificially high. Suppose, for instance, that an injector has become partly blocked. The fuel being consumed by the engine will be less, because of the blockage, whilst the fuel supply pressure will increase because of it. Consequently, the fuel flow indication would be falsely high. To measure actual flow rates it is necessary to place a device in the fuel supply line to the engine which will convert fuel flow into mechanical movement and transmit a signal to a fuel flow gauge in the cockpit. A simple type of flow measuring device, used in some piston engine and smaller gas turbine engine aircraft, is shown in Figure 7.17.

Fuel is directed through a volute chamber in which there is a spring-loaded vane. The action of the fuel pressing on the flat vane causes it to rotate

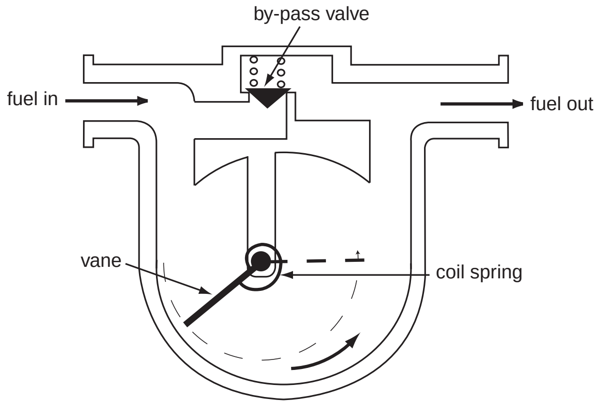

Figure 7.17 Vane-type flowmeter.

against the force of a coil spring until the gap between the edge of the vane and the inside of the volute chamber no longer impedes fuel flow. The rotation of the vane is transmitted to a fuel flow gauge in the cockpit by means of a synchro system. The greater the fuel flow, the more the vane will be deflected against the force of the coil spring before the gap is sufficiently wide. To protect against failure of the vane or other blockage in the volute chamber, a lightly loaded by-pass valve will open in the event of the differential pressure across the chamber exceeding a preset value, thereby maintaining fuel flow to the engine.

Integrated flowmeter

Larger, long-range, turbine-powered transports employ a more complex type of fuel flowmeter that is much more accurate than the foregoing systems and which integrates the measured flow rate with time, either mechanically or electronically, to compute and indicate fuel consumed as well as fuel flow. An example of the fuel flow measuring system is shown in Figure 7.18. The flowmeter is situated in the high pressure fuel supply to the engine and contains an impeller driven at constant rotary speed by an a.c. synchronous motor. Fuel entering the flowmeter passes through the impeller, which imparts a swirling motion to the fuel flow. The flow then enters a turbine where the force of the swirling fuel striking the turbine vanes drives the turbine to rotate in the same direction as the impeller rotation. The rotary motion of the turbine is restrained by a coil spring attached to the turbine shaft. As flow rate increases, the force of the swirling fuel acting on the turbine vanes increases, rotating the turbine further against the spring.

The rotary motion of the turbine shaft is sensed by a low voltage trans-

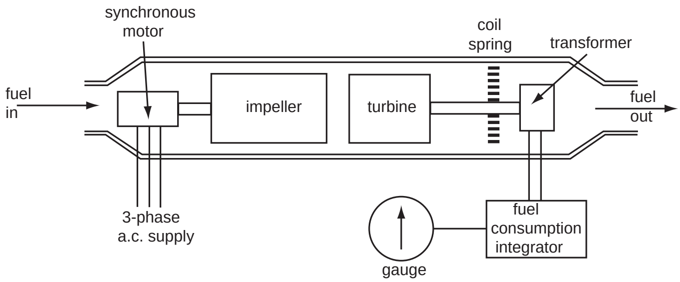

Figure 7.18 Integrated flowmeter.

former (LVDT), which produces an output voltage proportional to the amount of rotation. This output voltage forms the signal transmitted to the fuel consumption display, which is typically a digital readout in kilograms or pounds of fuel consumed. The signal also operates a servo-motor, which positions the pointer of an analogue fuel flow gauge, showing fuel flow in kilograms or pounds per hour. The two displays are often on the same instrument and an example is shown in Figure 7.19.

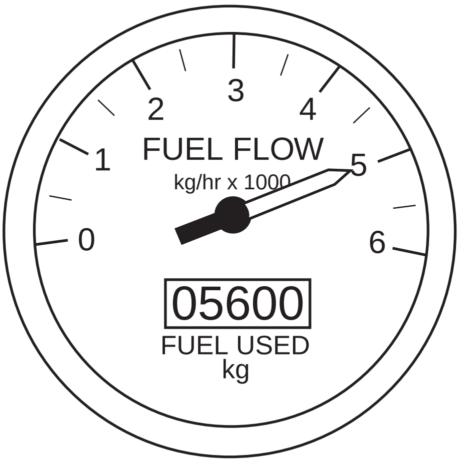

Figure 7.19 Combined fuel flow and consumption gauge.

Failure of the integrated flowmeter impeller or turbine will not impede the fuel flow to the engine, but will obviously render the gauge readings useless. Failure is typically indicated by a warning flag on the instrument display. The flowmeter circuitry may also incorporate a low fuel flow warning.

Fuel quantity measurement

In addition to knowing the rate at which fuel is being consumed, and the cumulative total of fuel consumed, it is clearly essential that the pilots are also aware of the quantity of fuel remaining in the aircraft tanks and so some form of tank gauging is necessary. One of the simplest forms of aircraft fuel tank gauge ever devised comprised a float-operated vertical dipstick, which protruded through a hole in the top of the tank at a location visible from the cockpit. The extent of dipstick protruding directly indicated the quantity of fuel remaining in the tank.

Resistive fuel quantity measurement

Such a system as that described above is obviously impractical for a large transport aircraft where the pilots and the automated systems need to know the exact quantity of fuel in each tank. However, in smaller aircraft a system of remote tank contents indication is often employed which does not differ greatly in principle from the float-operated dipstick. A float in the tank operates the wiper of a potentiometer, the output of which drives a moving coil instrument. As fuel level in the tank varies, the float moves up or down accordingly and operates the potentiometer, which is supplied with low voltage from the aircraft electrical system. The potentiometer signal causes the moving coil in the indicating gauge to position itself according to the strength of the signal, positioning the gauge pointer against a calibrated scale to show tank contents in gallons or litres. The principle of this method of fuel quantity measurement is illustrated in Figure 7.20.

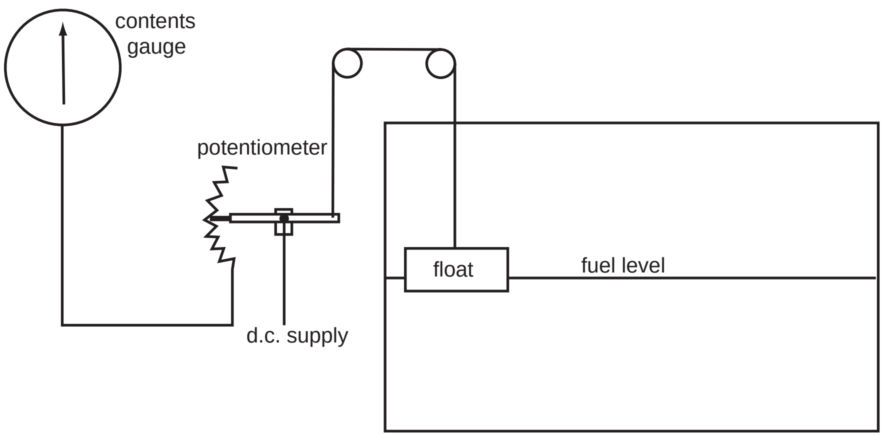

Figure 7.20 Resistive tank contents measurement.

This system of measuring the amount of fuel remaining in a tank suffers from a number of potential inaccuracies and is only suitable for use in the relatively small fuel tanks of light general aviation aircraft. Movement of fuel in the tank, due to changes of aircraft attitude or acceleration, will cause the float to move up or down and the system will falsely indicate the fuel quantity in the tank. Furthermore, since the system can only measure the *level* of fuel in the tank, it will give a false indication of quantity if the level changes due to temperature change. As both of these causes of inaccuracy are limited in the small tanks of light aircraft, they are generally acceptable. Additionally, the system is only capable of indicating the volume of fuel contained in the tank, computed from the fuel level and the known tank dimensions, so the associated fuel tank contents gauge must be calibrated in litres or gallons, rather than the preferred kilograms or pounds.

Capacitive fuel quantity measurement

The fuel tank contents in large aircraft, with a significant number of correspondingly large tanks, are measured by a system that uses the electrical capacitance of the fuel to determine the exact quantity of fuel in each tank and to indicate it in terms of mass rather than volume.

A capacitor is an electrical device consisting basically of two conducting plates separated by a resistive medium known as a dielectric. Such a device is capable of storing an electric charge and this property is known as capacitance. The principle of operation of a capacitor is shown in Figure 7.21.

When the two conducting plates of the capacitor are connected to an electrical supply, a potential difference exists between the plates. In the example in Figure 7.21 the switch in the circuit has been placed in position A and a 12 V supply has been connected to the plates. Current cannot flow between the plates since they are separated by a non-conducting dielectric, and so the potential difference between the plates is 12 V. If the switch is now moved to position B, the potential difference will continue to exist between the plates, since there is nowhere for it to discharge to. If the switch is now moved to position C the capacitor will discharge its stored electrons through the circuit provided.

In this example direct current is used and it will be noted that the capacitor only becomes charged, or discharges, when the switch is operated. If, however, a capacitor is supplied with alternating current, the constantly changing voltage and polarity of the supply acts in a similar manner to the switch. When voltage is increasing, the capacitor is charging as the potential difference between the plates increases. As the supply voltage decreases, the potential difference between the plates becomes greater than the supply voltage and the capacitor discharges. This process is continuously repeated through each half cycle of the a.c. supply.

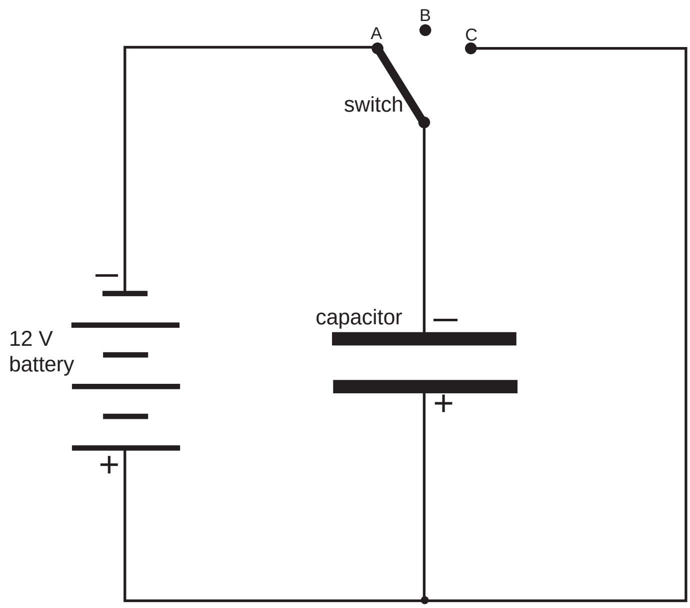

Figure 7.21 Capacitor principle.

Capacitance, the ability to store an electric charge, depends upon the surface area of the capacitor plates and the permittivity of the medium separating the plates. Permittivity is also sometimes referred to as the dielectric constant of the separating medium. It is usually measured as a relative value, where air has a permittivity of 1.00, so other media are assigned a relative permittivity indicating the capacitance they offer relative to air. Aviation kerosene, for example, has a relative permittivity of 2.10.

The capacitance or 'charge-holding capability' of a capacitor is the ratio between the charge supplied and the potential difference between the two plates and is measured in picofarads (pF). The strength of the discharge current from a capacitor will depend upon its capacitance and the rate of change of the supply voltage. The latter is a constant given a constant frequency a.c. supply and the capacitance, as we have seen, is dependent upon the permittivity of the medium. Suppose two identical capacitors are supplied with 6 V a.c. at a frequency of 400 Hz and one is placed in air whilst the other is placed in aviation kerosene. The capacitor with air separating its plates will produce a lower discharge current than the one with fuel separating its plates.

The capacitive probe inserted into an aircraft fuel tank is basically two open-ended tubes, one inside the other. The tubes act as the capacitor 'plates', separated by air or fuel, depending upon the depth of fuel in the tank. The concept is illustrated in Figure 7.22.

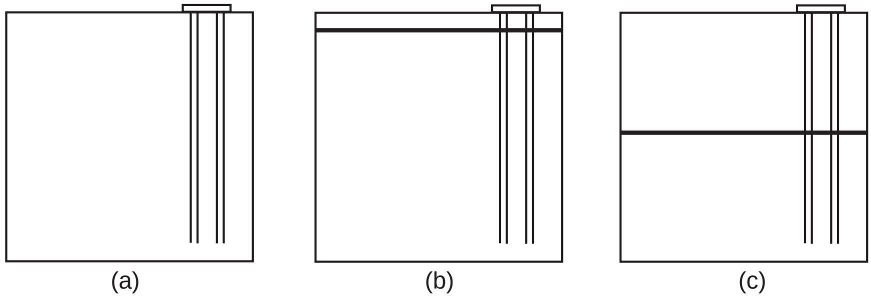

Figure 7.22 Effect of fuel level on capacitance.

In Figure 7.22(a) the tank is empty and the separating medium is air, with a relative permittivity of 1.00. For the sake of simplicity, let us assume that the capacitance of the probe is 100 pF. In Figure 7.22(b) the tank is filled with aviation kerosene with a permittivity of 2.10, so the capacitance of the probe will now be 210 pF, an increase of 110 pF, and its discharge current, when it is supplied with constant frequency a.c., will be 2.10 times greater than when the tank was empty. If the tank is half full, as shown in Figure 7.22(c), the increase in capacitance over the 'tank empty' value will be half as much, i.e. 55 pF. Thus, the capacitance of the probe will be 155 pF and the discharge current will be 1.55 times the empty value. From this it can be seen that the tank fuel level is accurately represented by the probe discharge current, and this is used to operate the tank contents gauge.

Compensation for movement of fuel in the tank due to aircraft attitude or acceleration changes is easily made by inserting several probes at different locations within the tank and 'averaging' their outputs to give a mean reading. Smaller tanks typically have two probes, whereas larger ones may have as many as six.

A simplified capacitive tank contents measuring system is shown in Figure 7.23. The low voltage alternating current supply to the system is from a power transformer to the fuel tank capacitor probe and to a reference capacitor in parallel with it. Because the fuel tank should never be completely empty, the charge from the probe will always be greater than the constant value of the reference capacitor. The higher the fuel level in the tank, the greater the difference and it is this potential difference that actuates the voltmeter, which is calibrated to read tank contents. A complete capacitive system is more complex than this, since it contains circuitry that allows the indicator gauge to be 'zeroed', that is the maximum and minimum readings to be adjusted for purposes of calibration.

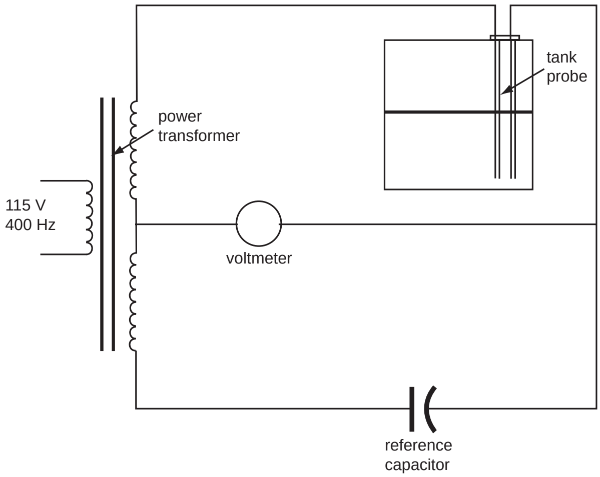

Figure 7.23 Simplified capacitive gauging system.

Fuel quantity by weight

The observant reader will have noted that, so far, all references have been to the measurement of fuel level in the tank, but it is ideally required that the measurement of fuel contents should be by weight. We have already established that fuel will expand as its temperature increases and the level of fuel in the tank will increase, whilst the important factor, its weight, remains the same. Therefore, if the gauges are calibrated to show quantity by weight, one would expect them to be in error.

However, the increase in volume with no change in weight means, by definition, that the fuel density has decreased. The reduction in density reduces the permittivity of the fuel and so the capacitance of the probe is reduced. This reduction almost exactly mirrors the increase in capacitance due to the higher level of fuel and the two effectively cancel each other out. Thus, the capacitance method of fuel tank measurement automatically compensates for changes in fuel density due to temperature changes.

Torque meter

The function of the torque meter is to measure and indicate the power developed by a turbo-propeller engine. The turning moment, or torque, delivered to the propeller through the reduction gearing is proportional to the horsepower developed, which is the product of the torque and the propeller rpm.

In the example of a torque meter system shown in Figure 7.24, the reduction gearbox between the engine and the propeller uses a type of

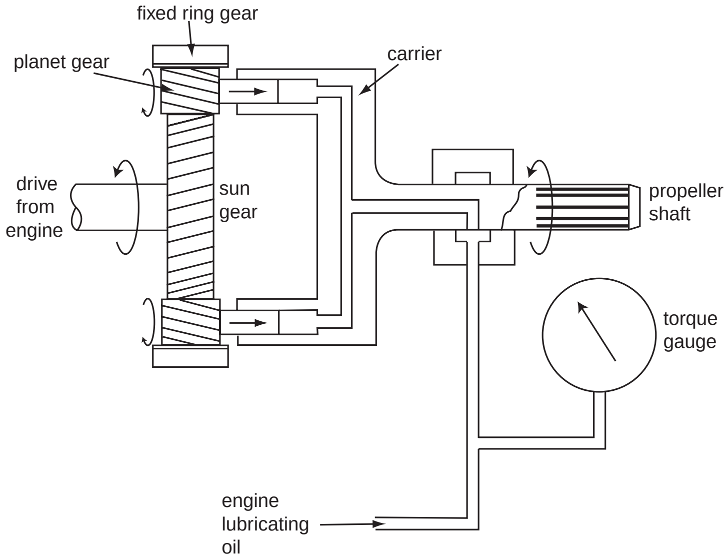

Figure 7.24 Torque meter system.

gearing known as sun and planet, or epicyclic. A large gearwheel, with helical teeth, drives smaller helical-toothed gearwheels surrounding it. These are the sun and planet wheels from which the name of the system is derived. The planet wheels rotate within a fixed ring gear and are carried in a drum attached to the propeller shaft, thereby transmitting rotation to the propeller shaft.

The torsional force of the engine turning the propeller is transmitted through the planet gears, the helical teeth of which transfer some of that force into an axial direction. The shafts of the planet gears form pistons that fit into cylinders machined in the carrier drum. These cylinders are connected to an enclosed hydraulic system, supplied with oil from the engine lubricating system and connected to a pressure gauge. Axial movement of the sun gears is restrained by the hydraulic system, but the greater the torsional force, and consequent axial force, the greater the hydraulic pressure created. The gauge reading is usually calibrated to read torque in ft lb and can be used to calculate the brake horsepower (bhp) being delivered to the propeller using the formula bhp = TNK, where T is the torque in ft lb, N the engine rpm and K a constant ($2\pi/33\,000$).

In some systems the gauge may also show negative torque, the undesirable condition that exists when the propeller is windmilling and tending to drive the engine. The torque meter output is used in some turbo-propeller systems to automatically operate the propeller feathering mechanism in the event of engine failure and excessive negative torque.

An alternative type of torque measuring system makes use of the fact that, under torsional loading, the propeller drive shaft twists. Strain gauges attached to a torque ring mounted on the engine/propeller drive shaft twist and deform with the shaft as torque increases. These produce a small electrical signal that varies according to the deformation of the strain gauges and is amplified to actuate the torque meter on the flight deck.

The torque meter gauge scale may incorporate coloured arcs. A green arc indicates the normal operating range of positive torque, a yellow arc indicates negative torque and red radial lines or arcs indicate maximum and minimum torque limits or ranges.

Vibration monitoring

Unlike piston engines, gas turbines have no reciprocating parts and the rotating assemblies are finely balanced dynamically. Consequently, they are much less prone to vibration under normal circumstances and any abnormal vibration is a clear indication of loss of dynamic balance due to damage. This may be caused by factors such as erosion, distortion or chipping of turbine blades, or ingestion damage to fan or compressor blades.

The vibration sensor comprises a permanent magnet suspended on springs and mounted on the engine and/or fan casing such that it is sensitive to radial oscillations. A pick-off coil surrounding the permanent magnet is connected through suitable circuitry to a vibration indicator and a warning circuit. The principle of operation is illustrated schematically in Figure 7.25.

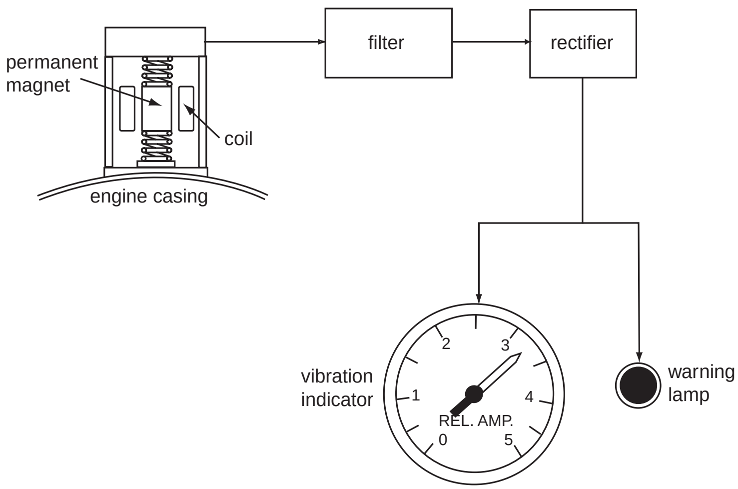

Figure 7.25 Vibration monitor.

Significant radial vibration or oscillation of the casing will only occur if there is a loss of dynamic balance in the adjacent rotating assembly. Typical locations for the vibration sensor are on the HP turbine casing and the fan casing on turbo-fan engines. As the casing and the attached sensor vibrate radially, the spring-suspended magnet remains virtually stationary. Thus, the coil of the sensor unit oscillates rapidly relative to the permanent magnet. The magnetic field is therefore cut by the coil, inducing a voltage in the coil proportional to the rate and extent of vibration.

The voltage output of the coil is amplified and integrated by a system of filters to eliminate known normal vibrations and to isolate vibration frequencies associated with particular components such as turbines, compressors, gearing, etc. The signal is then rectified and supplied to the vibration indicator on the flight deck, which indicates vibration in units of amplitude relative to a fixed datum value. The signal also feeds a warning circuit that illuminates a warning light when vibration exceeds a predetermined value.

Engine vibration is measured in units of *relative amplitude*.

More recently developed vibration monitoring systems employ piezoelectric sensors in conjunction with CRT displays such as EICAS and ECAM.

Remote signal transmission systems

The flying control surfaces and landing gear systems are not visible from the cockpit of most aircraft, but knowledge of their positions, particularly secondary control surfaces and landing gear, is important to the flight crew.

In some light aircraft the position of the landing gear and flaps is indicated to the pilot through a mechanical system of cables, push-pull rods and linkages. In most aircraft, however, transmission of position indicating signals is electrical.

For simple up, down, locked or unlocked indications the cockpit presentation may be in the form of lights or 'doll's eye' captions actuated by microswitches in the landing gear and flap actuating mechanisms. More complete indications showing the full range of movement of primary or secondary flying control surfaces may be transmitted by synchronous transmission systems of the type illustrated in Figure 7.3 or induction transmitters as shown in Figure 7.4.

Remote control

Remote control of primary and secondary flight controls and landing gear may be mechanical, electrical, hydraulic or a combination of all three. In light

general aviation aircraft, remote control transmission systems are usually mechanical, using push-pull rods, chain drives, cable-and-pulley systems and linkages between the pilot's controls and the systems to be operated. Larger aircraft typically use hydraulic transmission of movement from the pilot's controls to hydro-mechanical servo-actuators. In many large modern transport aircraft the remote transmission of signals from the pilot's controls to the hydro-mechanical servo-actuators is electrical.

Direct mechanical remote control systems have the advantage of simplicity and ease of maintenance, but are unsuitable for use in larger aircraft where the control loads are too great for direct manual force. To provide the power necessary to overcome the aerodynamic loads of large control surfaces and the weight of heavy landing gear, high pressure hydraulic actuators are required. The principal disadvantage with hydraulic systems is the requirement for absolutely pressure-tight systems and the fire and corrosion hazards associated with fluid leakages. Electrical transmission of remote control signals largely overcomes these disadvantages, although hydraulic actuators are still needed to move the control surfaces and landing gear. Electrical transmission has the added advantage of direct compatibility with computerised and automatic control systems.

Sample questions

1. A hydraulic system operates at a pressure of 205 bar. This pressure is equivalent to:

   a. 3015 psi?
   b. 2050 psi?
   c. 310 kg/cm^2?
   d. 250 kg/cm^2?

2. The pressure sensing element of a manifold pressure gauge typically comprises:

   a. A Bourdon tube?
   b. A single aneroid capsule?
   c. A single diaphragm?
   d. Two capsules, one pressure and one evacuated?

3. A synchronous transmission system requires:

   a. A d.c. supply?
   b. An a.c. supply?
   c. No electrical supply since it is self-generating?
   d. A synchronous motor at the transmitting element?

4. A potentiometer:

   a. Is a solid state electronic device?
   b. Is an electro-magnetic instrument?
   c. Is a rotary variable resistance device?
   d. Is used to measure potential difference?

5. The blue arc on a manifold air pressure gauge indicates:

   a. The rich mixture range?
   b. The take-off power range?
   c. The lean mixture range?
   d. The icing hazard range?

6. The resistive type of temperature gauge operates on the principle that:

   a. The resistance of a conductor decreases indirectly with temperature?
   b. The resistance of a conductor increases directly with temperature?
   c. Electrons will travel along a conductor from hot to cold?
   d. Electrons will travel along a conductor from cold to hot?

7. A thermocouple:

   a. Requires an a.c. supply?
   b. Requires a d.c. supply?
   c. Operates on the principle that conductivity varies with temperature?
   d. Requires no external electrical supply?

8. The cylinder head temperature of a piston engine is typically measured using:

   a. A surface contact thermocouple?
   b. An immersion thermocouple?
   c. A resistive temperature measurement system?
   d. A capacitive temperature measurement system?

9. The yellow arc on a carburettor air intake temperature gauge indicates:

   a. The icing hazard range of temperatures?
   b. The detonation hazard range of temperatures?
   c. The normal operating range of temperatures?
   d. The approach power range of temperatures?

10. An electrical tachometer system employs:

    a. A two-phase a.c. generator and motor?
    b. A three-phase a.c. generator and synchronous motor?

c. A d.c. generator and motor?
d. A synchro transmission system and moving coil indicator?

11. When a gas turbine tachometer is indicating 100% rpm, this indicates:

a. That the engine is operating at maximum rpm?
b. That the engine is operating at normal rpm?
c. That the engine is operating at take-off power?
d. That the engine is operating at optimum rpm?

12. A tacho-probe rpm measurement system:

a. Requires an a.c. electrical supply?
b. Requires a d.c. electrical supply?
c. Requires no external electrical supply?
d. Is only suitable for use with piston engines?

13. The impeller of an integrated flow meter:

a. Is driven by the turbine?
b. Is driven by an a.c. synchronous motor?
c. Is driven by the action of the fuel flow?
d. Drives the turbine at constant speed?

14. A resistive fuel quantity gauging system:

a. Uses a potentiometer to convert fuel level into an electrical signal?
b. Uses a capacitor to convert fuel level into an electrical signal?
c. Typically displays fuel quantity in terms of weight?
d. Automatically compensates for changes in fuel density due to temperature?

15. The permittivity of aviation kerosene is:

a. 1.0?
b. 1.55?
c. 0.55?
d. 2.1?

16. A capacitive fuel quantity gauging system is able to measure tank contents by weight because:

a. The permittivity of fuel varies directly with its density?
b. There are a number of capacitive probes in each tank?
c. The permittivity of fuel remains constant regardless of temperature?
d. The density of fuel remains constant regardless of temperature?

Answers to Sample Questions

Chapter 1: Air Data Instruments

Questions	Answers
1.	b
2.	a
3.	c
4.	b
5.	d
6.	c
7.	a
8.	d
9.	b
10.	d
11.	c
12.	a

Chapter 2: Gyroscopic Instruments and Compasses

Questions	Answers
1.	b
2.	c
3.	a
4.	d
5.	b
6.	d
7.	a
8.	c
9.	b
10.	d
11.	c
12.	a
13.	d
14.	b
15.	c
16.	c
17.	a
18.	b
19.	b
20.	c
21.	c
22.	a
23.	c

Chapter 3: Inertial Navigation Systems

Questions	Answers
1.	b
2.	a
3.	d
4.	c
5.	a
6.	c
7.	b
8.	d
9.	a
10.	c
11.	b
12.	d

Chapter 4: Electronic Instrumentation

Questions	Answers
1.	b
2.	b
3.	d
4.	a
5.	c
6.	c
7.	a
8.	b
9.	c
10.	d
11.	b
12.	d
13.	a
14.	c
15.	d
16.	a
17.	c
18.	c

Chapter 5: Automatic Flight Control

Questions	Answers
1.	b
2.	a
3.	d
4.	a
5.	c
6.	d
7.	c
8.	b
9.	a
10.	b
11.	c
12.	d
13.	d
14.	c
15.	b
16.	a

Chapter 6: In-Flight Protection Systems

Questions	Answers
1.	a
2.	b
3.	c
4.	c
5.	d
6.	b
7.	c
8.	a
9.	a
10.	b
11.	c
12.	a
13.	d
14.	a
15.	c
16.	b
17.	c
18.	d
19.	d
20.	b
21.	a
22.	b
23.	c

Chapter 7: Powerplant and System Monitoring Instruments

Questions	Answers
1.	a
2.	d
3.	b
4.	c
5.	c
6.	b
7.	d
8.	a
9.	a
10.	b
11.	d
12.	c
13.	b
14.	a
15.	d
16.	a

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